ATS SPACECRAFT POWER SYSTEM CONFIGURATION STUDY

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September 1970
Final Report for Period 1 January 1970 through 31 August 1970

Prepared For
GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland 20771

TECHNICAL REPORT STANDARD TITLE PAGE

1 Report No.	2 Government Accession No	3 Recipient's Catalog No
4 Title and Subtitle ATS SPACECRAFT SYSTEM CONFIGURATION STUDY		5. Report Date September 1970
		6. Performing Organization Code
7 Author(s)		8 Performing Organization Report No. SSD 00428
9. Performing Organization Name and Address Hughes Aircraft Company, Space Systems Division 1950 E. Imperial Highway El Segundo, California 90009		10 Work Unit No
		11 Contract or Grant No NAS 5-21123
12 Sponsoring Agency Name and Address National Aeronautics and Space Administration, Goddard Space Flight Center, Greenbelt, Maryland 20771 W. M. Tucker		13 Type of Report and Period Covered Final Report Jan. 1, 1970 to Aug. 31, 1970
		14 Sponsoring Agency Code
15. Supplementary Notes		

16 Abstract

The object of this report is to present data which will aid in optimization of regulation and conversion of spacecraft electrical power. The study is based on the applications technology satellite mission requirements and power system designs The ATS power systems use a decentralized bus with remotely located regulators and converters. A detailed analysis and discussion of the existing systems is provided with unit level data tabulation of size, weight, parts count, efficiency, reliability, and electrical performance. Three competing centralized systems are designed to provide equivalent electrical performance Similar details of these designs are tabulated for comparison with the present decentralized design. The major changes occur when comparing a centralized DC regulator design to the present decentralized design The centralized regulator design uses fewer parts, and has improved reliability and higher load power at the beginning of life, although it is slightly heavier.

Adding a centralized converter to the DC centralized regulation system had negligible additional impact. Use of a centralized AC distribution system is shown to be heavier and less efficient than other centralized systems. The study concludes with an examination and tabulated discussion of all factors which have a bearing on the power system design and centralization decision.

17 Key Words (Selected by Author(s)) Power Systems configurations, Applications Technology Satellite power systems. Centralized versus decentralized power systems.	18 Distribution Statement

19 Security Classif (of this report)	20 Security Classif (of this page)	21 No of Pages	22 Price*
UNCLASSIFIED	UNCLASSIFIED	188	

*For sale by the Clearinghouse for Federal Scientific and Technical Information, Springfield, Virginia 22151.

FOREWORD

This report has been prepared by the Power Systems Department of the Space Systems Division of Hughes Aircraft Company, El Segundo, California for NASA-Goddard Space Flight Center, under Contract NAS 5-21123.

Technical direction and guidance during this study has been provided by Mr. W.M. Tucker, NASA technical officer on this contract and Mr. C.M. MacKenzie, NASA GSFC.

Technical direction at Hughes was the responsibility of Mr. E. Levy, Jr. with principal contributions by Mr. S. Sung and Mr. W.E. Michel. ATS systems engineering support was provided by Mr. P. Sengstock and reliability analysis by Mr. L. Cole.

A draft of the final report was reviewed by Messrs. J. Radecki, P. Fono, and P.S. DuPont and this report incorporates their suggestions and comments.

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1. INTRODUCTION AND SUMMARY

The object of this report is to present data which will aid in the optimization of regulation and conversion of power in a spacecraft electrical power subsystem. The basic consideration is the extent to which regulation and/or conversion in a power system should be centralized. The study is based primarily on the Applications Technology Satellite mission requirements and spacecraft power systems designs.

The Applications Technology Satellite (ATS) provides a relatively large, adaptable payload capability designed to achieve long life in circular, medium altitude orbits or synchronous, equatorial orbit. Gravity gradient stabilization maintains vehicle orientation in the medium altitude orbit and either gravity or spin stabilization, depending on mission objectives, is used for vehicle orientation in the synchronous orbit.

The first of five spacecraft was launched during the second half of 1966. Subsequent launches occurred approximately every 6 months.

The ATS power system uses a decentralized bus concept which resulted from design tradeoffs at the outset of the program. At that time, however, most of the loads were in the preliminary design phase and many assumptions had to be made in order to make timely decisions and provide flexibility. With load data and spacecraft constraints established, NASA-GSFC recognized the desirability of determining whether an optimized regulated or unregulated, centralized or remote system is more desirable for each set of ATS circumstances. As a result, this study was initiated.

In order to establish the advantages and disadvantages of the various possible power system configurations, it was necessary initially to study the existing ATS and to understand the existing power system design, remote regulator converter design, load characteristics, and other spacecraft requirements. This study of the present system resulted in sufficient data to conceive and design various centralized systems which could provide power to all the ATS loads with adequate regulation, filtering, protection, and other needed characteristics. It was determined that certain loads presently using remote regulator converters (the TWT tubes, for example) did not lend themselves to the use of centralized power. Consequently, for the centralized system design, these loads retained the regulator converters as used in the present system.

A number of centralized regulator and regulator converter systems were considered. Three systems were selected for design and comparison with the present ATS power system. Design features, failure modes, component stress levels, and all aspects of the concepted designs were considered in the same depth as a flight system in order that meaningful comparisons could be made. The three systems designed were a centralized regulation system, a centralized regulation system with a centralized converter, and an ac distribution system.

Table 1-1 summarizes the major comparisons between a typical ATS power system (ATS-B) and the three centralized systems. The major difference between systems occurs in the change from the ATS-B system to the centralized regulation system. The centralized regulation system uses fewer parts while achieving substantially higher reliability. These two features are the major advantages of the centralized system. An added advantage is improved efficiency at the beginning of life when the solar panel is operating at its maximum power point. Weight of the centralized regulation system is slightly higher and volume is approximately the same. Cost and performance could not be compared quantitatively but a qualitative analysis showed no major advantages for either system.

TABLE 1-1. CENTRALIZED POWER SYSTEM SUMMARY DATA

Parameter	POWER SUBSYSTEM			
	ATS-B	Centralized Regulation	Centralized Regulation and Conversion	AC Distribution System
Parts count	2461	1988	1990	2185
Reliability	0.870	0.975	0.975	--
Efficiency (beginning of life)				
Power out	129.4	129.4	129.4	129.4
Loss	38.19	27.76	26.29	38.79
Efficiency	77.2	82.3	83.0	77.0
Efficiency (end of life)				
Power out	118.55	118.55	118.55	--
Loss	25.74	25.29	24.45	
Efficiency	82.2	82.4	82.9	
Weight	22.54	25.99	26.27	29.09
Volume	783.9	787	792.6	908.1

Adding a centralized converter to the centralized regulation system had negligible impact on the system comparison. Only a small number of converters were used on ATS-B and the new design required replacement of six remote converters with two redundant centralized converters. However, even if more remote converters had been required in the ATS-B, very little advantage would result from use of centralized conversion.

The ac distribution system loses the efficiency advantage of the other two systems and is substantially heavier. Its only advantage is the use of fewer parts than the present system.

The above comparison is specific to the ATS designs. In order to expand the conclusions to other spacecraft, it is necessary to examine all factors that have a bearing on the centralization decision. These factors are listed in Table 1-2 and discussed in detail in this report. The major factors that appear to affect the ATS centralization decision are the number of loads, regulation required by the loads, and desirability of having additional power at the beginning of life. If only a few spacecraft loads are needed, then the redundant centralized system is overly complex. If precision regulation is needed in a high percentage of loads, remote regulators will be needed for these loads. For any particular application, all factors must be reviewed in order to determine if a centralized or decentralized system is advantageous.

TABLE 1-2. SUMMARY OF FACTORS AFFECTING CENTRALIZATION DECISION

Factors Affecting Centralization Decision	Major Importance to ATS Centralization Decision: Centralized Regulation
Load requirements	
Precision regulation	X
Number of voltage levels	
Current levels	
Transients	
EMI/ripple	
On/off - overload	
Number of loads	X
Load sequencing	X
Solar panel sizing	
Energy storage sizing	
Charge configuration	
Distribution system	
Spacecraft configuration	
Stabilization method	
Thermal control	
Launch vehicle	
Launch and orbit	
Cost and reliability	
Radiation	
Mission objectives and specification requirements	
Interface requirements	

2. REQUIREMENTS

This section tabulates the power subsystem requirements for the five ATS spacecraft. The requirements are listed in accordance with the design factors listed in the introduction.

Table 2-1 lists the loads for each spacecraft according to the spacecraft and the bus the load is required to be connected to. This includes the two main buses and the two battery buses. Loads requiring on/off control and overload shutoff are indicated. Loads 1 through 10 and 14a are part of the basic spacecraft. The other loads are the experiments. Many of the 1 through 10 loads are essential and are therefore supplied redundantly and connected to each bus to prevent complete loss of its function.

Table 2-2 lists all of the other ATS requirements. Many of the requirements are seen to be the same for all five spacecraft. Differences are shown where applicable. The data as shown in the tabulation is intended to summarize typically important features rather than precisely cover all aspects of the particular requirements.

Load sequencing is not shown in the requirements table since it is not a constraint on the ATS power system design. The launch load is minimal (300 ma per bus) and the panel power is available shortly after launch (< 5 minutes). Eclipse operation and load requirements before final orbit is reached are also not restrictive with respect to power. In final orbit there is sufficient power to run the desired experiments.

TABLE 2-1. SPACECRAFT LOADS

ATS-B			
	Bus 1		Bus 2
1a	Voltage limiters (4 required) Limits to -32 volts	1b	Voltage limiters (4 required) Limits to -32 volts
2a	Solenoid drivers (2 required) Pulse load - no regulation	2b	Solenoid drivers (2 required) Pulse load - no regulation
3a	Repeater regulator* FT: -24 volts 200 ma in 194 ma out MA: -24 volts 200 ma in 194 ma out WBDM: -24 volts 200 ma in 194 ma out	3b	Repeater regulator* FT: -24 volts 200 ma in 194 ma out MA: -24 volts 200 ma in 194 ma out WBDM: -24 volts 200 ma in 194 ma out
4a	TWT power supply (2 required) series regulator* -24 volts 750 ma in Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts - 800 volts (matched to TWT) -1400 volts	4b	TWT power supply (2 required) series regulator* -24 volts 750 ma in Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts - 800 volts (matched to TWT) -1400 volts
5a	PACE regulator* -24 volts 310 ma in 114 ma out Converter 180 ma in +24 volts 147 ma	5b	PACE regulator* -24 volts 310 ma in 114 ma out Converter 180 ma in +24 volts 147 ma
6	Phase shifter driver (2 required) regulator* -24 volts 300 ma in Converter		
7a	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma	7b	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma
8a	Telemetry transmitter* 2 -24 volts 180 ma in	8b	Telemetry transmitter* 2 -24 volts 180 ma in
9a	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma	9b	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma
10a	Accumulator regulator*	10b	Accumulator regulator*
11a	Cloud camera regulator* ⊗ -24 volts 600 ma (rated)	11b	EME regulator* ⊗ -24 volts 1460 ma (rated)
12a	Nutation experiment regulator* ⊗ -24 volts 1250 (rated)	12b	Ion engine regulator* ⊗ (clock off) -24 volts 750 ma (rated)
13a	VHF regulator* ⊗ (clock off) -24 volts 1900 ma (rated)	13b	VHF regulator* ⊗ (clock off) -24 volts 1900 ma (rated)
		20b	Strain gage amplifier

*Commandable on/off and overload shutoff.

⊗ Payload regulators - identical except for overload trip setting.

Table 2-1 (continued)

ATS-B (Continued)			
	Bus 1		**Bus 2**
	<u>Battery Bus 1</u>		<u>Battery Bus 2</u>
1c	Squib drivers (2 required) 4.5a each	1d	Squib drivers (2 required) 4.5a each
	ATS-A		
1a	Voltage limiters (6 required) Limits to -32 volts	1b	Voltage limiters (6 required) Limits to -32 volts
3a	Repeater regulator* FT: 24 volts 200 ma MA: 24 volts 200 ma WBDM: 24 volts 200 ma	3b	Repeater regulator* FT· 24 volts 200 ma MA: 24 volts 200 ma WBDM: 24 volts 200 ma
4a	TWT power supply (2 required) Series regulator* -24 volts 750 ma Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts (matched to TWT) - 800 volts -1400 volts	4b	TWT power supply (2 required) Series regulator* -24 volts 750 ma Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts (matched to TWT) - 800 volts -1400 volts
7a	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma	7b	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma
8a	Telemetry transmitter* (2 required) -24 volts 180 ma in	8b	Telemetry transmitter* (2 required) -24 volts 180 ma in
9a	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma	9b	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma
14a	Current control unit Set voltage limiters for 0.5 ampere per bus upon command.		
15a	Subliming solid driver* -24 volts	15b	Subliming solid driver* (2 required) -24 volts
16a	EME regulator* x -24 volts 1700 ma (rated)	16b	Albedo regulator* x -24 volts 1250 ma (rated)
17a	Met. regulator* x (2 required) -24 volts 1700 ma (rated)	17b	Gravity gradient regulator* x -24 volts 1250 ma (rated)
	<u>Battery Bus 1</u>		<u>Battery Bus 2</u>
2c	Payload power switch (on/off) (2 required) switches batteries to experiment	2d	Payload power switch (on/off) (2 required) switches batteries to experiment

*Commandable on/off and overload shutoff.

x Payload regulators - identical except for overload trip setting.

Table 2-1 (continued)

ATS-C			
Bus 1		Bus 2	
1a	Voltage limiters (4 required) Limits to -32 volts	1b	Voltage limiters (4 required) Limits to -32 volts
2a	Solenoid drivers (2 required) Pulse load - no regulation	2b	Solenoid drivers (2 required) Pulse load - no regulation
3a	Repeater regulator* FT: 24 volts 200 ma in MA: 24 volts 200 ma in WBDM: 24 volts 200 ma in	3b	Repeater regulator* FT: 24 volts 200 ma in MA: 24 volts 200 ma in WBDM: 24 volts 200 ma in
4a	TWT power supply (2 required) series regulator* -24 volts 750 ma in Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts (matched to TWT) - 800 volts -1400 volts	18b	TWT power supply (2 required) series regulator* -24 volts 1820 ma in Filament supply 4.5 volts ac 150 ma HV supply (commandable ON) (matched to TWT)
5a	MACE regulator* -24 volts 310 ma in 114 ma out Converter +24 volts 147 ma	5b	MACE regulator* -24 volts 310 ma in 114 ma out Converter +24 volts 147 ma
7a	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma	7b	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma
8a	Telemetry transmitter* (1 required) -24 volts 180 ma in	8b	Telemetry transmitter* (1 required) -24 volts 180 ma in
9a	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma	9b	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma
10a	Accumulator regulator*	10b	Accumulator regulator*
19a	Third harmonic generator* ⓧ -24 volts 750 ma (rated)	19b	Image dissector camera* ⓧ -24 volts 1000 ma (rated)
20a	Resistojet* ⓧ (clock off) -24 volts 750 ma (rated)		
21a	VHF* ⓧ (clock off) -24 volts 1900 ma (rated)	21b	VHF* ⓧ (clock off) -24 volts 1900 ma (rated)
22a	Self-contained navigation experiment* ⓧ -24 volts 600 ma (rated)	22b	Reflectometer* ⓧ -24 volts 1460 ma (rated)
23a	Multicolored spin-scan cloud camera* ⓧ -24 volts 100 ma (rated)		
24a	Mechanically despun antenna* ⓧ -24 volts 1460 ma (rated)	24b	Mechanically despun antenna* ⓧ -24 volts 1460 ma (rated)

*Commandable on/off and overload shutoff.

ⓧ Payload regulators - identical except for overload trip setting.

Table 2-1 (continued)

ATS-C (Continued)			
	Bus 1		Bus 2
	Battery Bus 1		Battery Bus 2
1c	Squib drivers 3 4.5a each	1d	Squib drivers 4.5a each
2c	Payload power switch (on-off) 1 switches batteries to experiment		
ATS-D			
	Bus 1		Bus 2
1a	Voltage limiters 6 limits to -32 volts	1b	Voltage limiters 6 limits to -32 volts
2a	Solenoid drivers 1 pulse load - no regulation	2b	Solenoid drivers 1 pulse load - no regulation
3a	Repeater regulator* FT: -24 volts 200 ma in MA: -24 volts 200 ma in WDBM: -24 volts 200 ma in	3b	Repeater regulator* FT: -24 volts 200 ma in MA: -24 volts 200 ma in WDBM: -24 volts 200 ma in
4a	TWT power supply 2 series regulator* -24 volts 750 ma Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts (matched to TWT) - 800 volts -1400 volts	4b	TWT power supply 2 series regulator* -24 volts 750 ma Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts (matched to TWT) - 800 volts -1400 volts
7a	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma	7b	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma
8a	Telemetry transmitter* 1 -24 volts 180 ma in	8b	Telemetry transmitter* 1 -24 volts 180 ma in
9a	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma	9b	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma
10a	Accumulator regulator*	10b	Accumulator regulator*
14a	Current control unit set voltage limiter for 0.5 ampere bus upon command		
15a	Subliming solid driver* -24 volts	15b	Subliming solid driver* 2 -24 volts

*Commandable on/off and overload shutoff.

ATS-D (Continued)			
	Bus 1		**Bus 2**
20a	Resistojet* ⊗ (clock off) -24 volts 750 ma (rated)		
12a	Ion engine experiment regulator* ⊗ -24 volts 750 ma (rated)	12b	Ion engine regulator* ⊗ (clock off) -24 volts 750 ma (rated)
		17b	Gravity gradient regulator* ⊗ -24 volts 1250 ma (rated)
25a	Image orthicon camera* ⊗ -24 volts 1500 ma (rated)	25b	Image orthicon camera* ⊗ -24 volts 2100 ma (rated)
26a	Magnetometer* ⊗ -24 volts 600 ma (rated)		
27a	Magnetic damper* ⊗ -24 volts 1700 ma (rated)		
	<u>Battery Bus 1</u>		<u>Battery Bus 2</u>
1c	Squib drivers 5 4.5 amperes each	1d	Squib drivers 5 4.5 amperes each
2c	Payload power switch (on/off) 2 switches batteries to experiment	2d	Payload power switch (on/off) 2 switches batteries to experiment
3c	Squib drivers 3 4.9 amperes each	3d	Squib drivers 3 4.9 amperes each
ATS-E			
	Bus 1		**Bus 2**
14a	Current control unit set voltage limiter for 0.5 amperes per bus upon command		
15a	Subliming solid driver* -24 volts	15b	Subliming solid driver* (2 required) -24 volts
		16a	EME regulator* ⊗ -24 volts 2100 ma (rated)
12a	Ion engine experiment regulator* ⊗ (clock off) -24 volts 750 ma (rated)	12b	Ion engine regulator* ⊗ (clock off) -24 volts 750 ma (rated)
20a	Resistojet* ⊗ (clock off) -24 volts 750 ma (rated)	20b	Strain gage amplifier
		17b	Gravity gradient regulator* ⊗ -24 volts 1250 ma (rated)
19a	Third harmonic generator* ⊗ -24 volts 750 ma (rated)		
26a	Magnetometer* ⊗ -24 volts 600 ma (rated)		
27a	Magnetic damper* ⊗ -24 volts 1700 ma (rated)		

*Commandable on/off and overload shutoff.

⊗ Payload regulators - identical except for overload trip setting.

Table 2-1 (continued)

ATS-E (Continued)			
Bus 1		**Bus 2**	
28a	Millimeter wave* (x) -24 volts 1700 ma (rated)	28b	Solar cell experiment* (x) -24 volts 600 ma (rated)
29a	Millimeter wave backup* (x) -24 volts 1000 ma (rated)		
	Battery Bus 1		Battery Bus 2
1c	Squib drivers (5 required) 4.5 amperes each	1d	Squib drivers (5 required) 4.5 amperes each
2c	Payload power switch (on/off) (2 required) switches batteries to experiment	2d	Payload power switch (on/off) (2 required) switches batteries to experiment
3c	Squib drivers (3 required) 4.9 amperes each	3d	Squib drivers (3 required) 4.9 amperes each
Bus 1		**Bus 2**	
1a	Voltage limiters (6 required) limits to -32 volts	1b	Voltage limiters (6 required) limits to -32 volts
2a	Solenoid drivers (1 required) pulse load - no regulation	2b	Solenoid drivers (1 required) pulse load - no regulation
3a	Repeater regulator* FT: 24 volts 200 ma in MA: 24 volts 200 ma in WDBM: 24 volts 200 ma in	3b	L-band repeater FT: 24 volts 200 ma in MA: 24 volts 200 ma in WDBM: 24 volts 200 ma in
4a	TWT power supply (2 required) series regulator* -24 volts 750 ma in Filament supply 4.5 volts ac 425 ma HV supply (commandable ON) + 200 volts - 800 volts (matched to TWT) -1400 volts	26b	TWT power supply (2 required) series regulator* -24 volts 2200 ma in Filament supply 4.5 volts ac HV supply (commandable ON) (matched to TWT)
7a	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma	7b	Telemetry encoder regulator SCO series regulator* -24 volts 15 ma Series regulator* -24 volts 146 ma in 62 ma out Converter 77 ma in -14 volts 49 ma +14 volts 25 ma +22 volts 24 ma
8a	Telemetry transmitter* (2 required) -24 volts 180 ma in	8b	Telemetry transmitter* (2 required) -24 volts 180 ma in
9a	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma	9b	Command regulator (overload protection) -24 volts 235 ma in 52 ma out Converter 149 ma in +24 volts 106 ma -5.5 volts 65 ma
10a	Accumulator regulator*	10b	Accumulator regulator*

*Commandable on/off and overload shutoff.

(x) Payload regulators - identical except for overload trip setting.

TABLE 2-2. ATS REQUIREMENTS

Requirement	SPACECRAFT A	B	C	D	E
SOLAR PANEL					
Configuration					
Each panel (2 required)					
Length, inches	24.2	26.5	26 5	24.2	24.2
Diameter, inches	56 0	57.6	57.6	56 0	56.0
Thickness, inches	0 52	0 64	0.64	0.52	0.52
Power Required					
Not including battery charge arrays					
Beginning of life	127 watts at -27 volts	165 watts at -27 volts	165 watts at -27 volts	127 watts at -27 volts	127 watts at -27 volts
End of life	75% of initial power after 3 years				
BATTERY					
Output Voltage	-24 5 volts minimum at 20% depth of discharge				
Capacity	6 amp-hr at 75°F and 1.2 amperes discharge rate 4 8 amp-hr at 100°F and 30°F and 1 2 amperes discharge rate				
High Current Discharge	1 0 volts per cell at 12 0 amperes for 10 seconds when fully charged				
Charge Efficiency	36%				
Cycle Life	10,000 cycles at 22% DOD	300 cycles at 22% DOD			
Repetitive Cycle	1 75 amp for 0 75 hour discharge, 5 65 hour charge	1 15 amp for 1 2 hour discharge, 22 8 hour charge			
DISTRIBUTION SYSTEM					
Main Bus					
Number of Buses	Two capable of being paralleled on command				
Max/Min Voltages	-32/-24 5 volts				
Ripple	500 mv peak-to-peak maximum				
Transients	Voltage must remain between -24 5 and -32 volts Transient duration 0 5 second or less				
Battery Bus (2)	Direct connection to each battery				

Table 2-2 (continued)

Requirement	SPACECRAFT A	B	C	D	E
SPACECRAFT CONFIGURATION					
Panel Mounting	Cylinders are mounted through holes in brackets attached to inside of panel. 45° sections are mounted by spring loaded ball assemblies				
Cutouts	Notch for damper boom	Cutouts for camera, jets, experiments →	→	Notch	Notch
Battery Mounting	Mounted to spacecraft structure typically without good conductive path.				
Electronics Mounting	High dissipation packages mounted using interface of RTV or indium foil No other mounting constraints.				
Shadowing	5%	None	None	5%	5%
STABILIZATION					
Type	Gravity Gradient	Spin Stabilization	Spin Stabilization	Gravity Gradient	Gravity Gradient
Spin Rate	1 rev/day	100 rpm	100 rpm	100 rpm for 2 weeks then 1 rev/day	
Jet Systems	2 inversion motors 2 E-W motors	2 radial 2 axial → 2 cold gas spinup jets	→	2 cold gas spinup jets → 1 yo-yo despin 1 axial 1 radial	→
THERMAL CONTROL					
Panel Temperature					
Maximum temperature/ description	150° when sun normal	85°F winter solstice →	→	150° when sun normal	60°F
Minimum temperature/ description	-80°F during eclipse or facing space	→	→	→	→
Seasonal bulk temperature					
Equinox	150°F at normal to -25°F at 90°	62°F	62°F	150°F at normal to 125°F at 90°	60°F
Summer solstice		50°F	50°F		
Winter solstice		70°F	70°F		
Battery Temperature					
Minimum on-charge	30°F →	→	→	→	→
Maximum overcharge	108°F	100°F	100°F	102°F	102°F
Soakback temperature	110°F →	→	→	→	→
Electronics					
Min/Max	30°/100°F →	→	→	→	→
Commandable Current Load	0 5 ampere per bus	--	--	0 5 ampere per bus	0 5 ampere per bus

Table 2-2 (continued)

Requirement	SPACECRAFT A	B	C	D	E
LAUNCH AND ORBIT					
Fairing Jettison					
Time	200 seconds				
Altitude	300, 000 feet				
Transfer Orbit					
Number	1-1/2 to 3-1/2				
Apogee	22, 752 miles				
Perigee	100				
Final Orbit					
Circular	6000 miles	22, 752 miles			
Inclination	28 5°	0	0	1.5	1.5
Maximum eclipse time	45 minutes	1 2 hours			
Minimum recharge time	5 65 hours	22 8 hours			
RELIABILITY					
Goal	3 year lifetime				
Features	1) Separate buses with redundant critical loads 2) Buses unparallel when fault reduces bus voltage. 3) Current limited through discharge control until buses unparallel. 4) Each load protected by overload control 5) Maximum voltage clamped				
RADIATION					
Synchronous Altitude					
Transfer eclipse		Particle Energy, electron volts	Integ. Flux, particles/cm^2	*	
Electrons		$>1\ 6 \times 10^6$ $>40 \times 10^3$	1×10^6 6×10^{12}		
Protons		$(0\ 1 \text{ to } 5) \times 10^6$ $>30 \times 10^6$	3×10^{12} 6×10^5		

*These values have been updated by data published after the ATS specification was written. This updated data is not included in this requirements section.

Requirement	SPACECRAFT A	B	C	D	E
Sync orbit (3 years)		Particle Energy, electron volts	Integ Flux, particles/cm^2 *		
Electrons		$>1\ 6 \times 10^6$ $>40 \times 10^3$	3×10^{16} 3×10^{15}		
Protons		$(0.1 \text{ to } 5) \times 10^6$ $>30 \times 10^6$	3×10^{15} 6×10^8		
Solar Flares	Protons greater than 12×10^6 e v has a total flux of 2×10^{12} protons/cm				
Medium Altitude 3 years	Particle Energy, electron volts / Integ Flux particles/cm^2				
Electrons	$1\ 6 \times 10^6$ / 3×10^{11} 50×10^3 / 2×10^{15}				
Protons	$(0\ 1 \text{ to } 5) \times 10^6$ / 8×10^{14} 30×10^6 / 5×10^9				
LAUNCH VEHICLE Vibration 4 35 min/axis	**Sinusoidal** 5 to 15 cps — 0.167 double amplitude 15 to 250 cps — 2.0 g peak 250 to 400 cps — 3 33 g peak 400 to 2000 cps — 5 0 g peak 4 35 min/axis		**Random** 20 to 80 cps 80 to 1280 cps 1280 to 2000 cps **Torsional** 50 to 80 cps	0 0178 g^2/cps increasing from 0.0178 g^2/cps at 0.61 dB/oct 0 0311 g^2/cps 5.0 peak tangential at 5 feet diameter	6 min/axis
ACCELERATION Type	20 rad/sec^2 Atlas-Agena			Atlas-Centaur	Atlas-Centaur

*These values have been updated by data published after the ATS specification was written This updated data is not included in this requirements section

Table 2-2 (continued)

Requirement	SPACECRAFT				
	A	B	C	D	E
INTERFACE					
Commands					
Execute signal					
Voltage	15 6 to 20 1 volts				
Impedance	7 73 to 10.5 kilohms				
Duration	50 ms minimum				
Telemetry					
Voltage	0 to -5 volts				
Impedance	150 kilohms				
Temperature sensor resistance	30 kilohms typical				
Status signal	0 = 0 volts 1 = -5.5 volts				

3. EXISTING ATS DESIGNS

This section provides a detailed description of the existing ATS power systems used on the five ATS spacecraft. The description includes the parameters of interest for the power configuration study. These parameters will then be used as a baseline for comparison with other competing systems.

The design philosophy underlying the ATS program was to develop a multiple mission satellite system allowing a choice of gravity gradient or spin attitude stabilization permitting inclusion of various experimental packages as payloads. To this end two different, but similar, primary structures were designed. One was tailored to the spin stabilization satellite mission and the other to the gravity gradient satellite mission. The items that figured significantly in the design process were a consideration of the thermal environment, the desire to achieve a near optimal design for a solar array, and the desire to provide payload mounting space with the maximum flexibility.

As a result of the above criteria, the ATS power system used a decentralized bus concept. An unregulated bus distributed power to the spacecraft load and experiments. Each spacecraft load and experiment was provided with its own regulator or regulator-converter.

Figure 3-1 is a composite functional block diagram of all ATS electrical systems and reflects the decentralized design philosophy. Electrical power for all three versions of the ATS spacecraft is provided by solar cell arrays and rechargeable nickel-cadmium batteries. The solar cells provide the primary power source of electrical energy to power the spacecraft load while the batteries are used to provide electrical power during transient loads and solar eclipse.

The solar arrays and batteries are divided into two separate bus systems to maximize reliability and operational flexibility. Each main solar array directly powers an unregulated spacecraft bus whose voltage range is maintained between -32 and -24.5 volts. The upper limit is controlled by the bus voltage limiters and the lower by the battery discharge control set point. Each battery is charged directly by a small solar cell battery charge array which provides the required charge voltage and also acts as a battery charge current limit. A bus relay is used in the spacecraft to tie the buses together to provide for asymmetrical spacraft

subsystem loads and experimental payloads. All ATS regulators are of the series dissipative type. Except for minor exceptions described under specific units, most outputs are -24 volts. Where other voltages are required, a conventional dc-dc converter is employed using the regulated -24 volts as input.

The composite block diagram shows the power subsystem, remote regulator converters, and loads for the various spacecraft. The dotted lines indicate loads that are on some of the spacecraft. The solid lines are loads that are on all spacecraft. An additional bus is brought out directly from each battery to power certain critical loads.

Figure 3-2 is a typical block diagram of half of the system illustrating the commands and voltage outputs of the various regulators and converters. Dotted lines in this diagram also indicate loads that are not on all spacecraft. The loads shown in the block diagrams are listed in Table 3-1 showing the units used on all five of the ATS spacecraft. The table lists the subsystem each unit is used on, the number of units required for each spacecraft, size, weight, function, regulation technique, power rating, and other special incorporated provisions. A typical spacecraft (ATS-B) uses 6 shunt regulators, 24 series regulators, and 12 dc to dc converters.

The centralized power system units shown on the block diagrams and listed in the table have been further analyzed to determine weight, volume, parts count and beginning of life and end of life output power, power dissipation, and efficiency. Tables 3-2 and 3-3 are summaries of this data.

Table 3-2 shows weight, volume, and parts count for each unit and for each spacecraft. Weight ranges from 18.7 pounds for ATS-C to 26.31 pounds for ATS-E. ATS-E also has the largest volume and highest parts count, due to several factors. The ATS-E is the only spacecraft that uses a switching converter (5.56 pounds and 205 parts); the switching converter is needed to lower the higher bus voltage introduced by the heat pipe experiment). ATS-E also uses six voltage limiters (versus four for some other spacecraft) and carries 11 experiments (versus a minimum of five) requiring 11 payload regulators.

Table 3-3 compares power output and losses for two spacecraft: ATS-A (medium altitude gravity gradient) and ATS-B (synchronous altitude spin stabilized) at both beginning of life and end of life. All comparisons are made by assuming that the solar panel is operating at its maximum power point which is 26.9 volts for ATS-B and 27.5 volts for ATS-A at the beginning of life and 25.5 volts at the end of life for both. The data shows efficiency improvement of about 5 percent from beginning of life to end of life. Also, as the load increases, efficiency improves because fixed losses become a smaller percentage of the total. The values arrived at in these summary tables will be used to compare to the centralized designs described in Section 5 of this report.

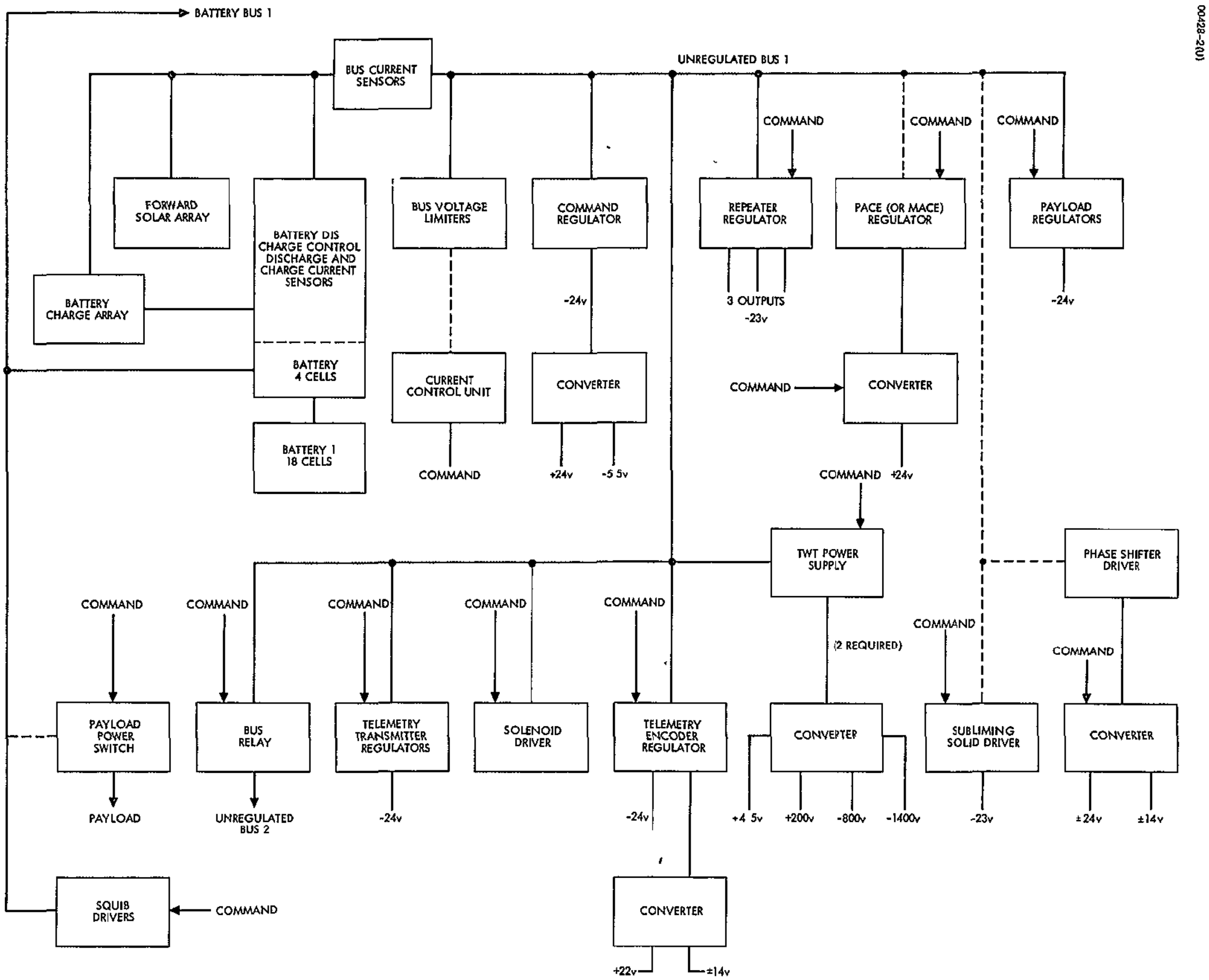

Figure 3-2. Typical Block Diagram of One-Half ATS Spacecraft System

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TABLE 3-1. EXISTING ATS DESIGNS.

Unit Name	Used On	Number Required for each Spacecraft B (F-1)	A (F-2)	C (F-3)	D (F-4)	E (F-5)	Unit Size, inches	Unit Weight, pounds	Functions	Regulation Technique	Unit Power Rating, watts	Other Provisions On-Off	Overload
Forward Solar Array (Battery Charge Array 1)	Power Subsystem	1	1	1	1	1	T = 0 64 L = 26 3 D = 57 6	31 08	Primary power source for bus 1 and battery charge power for battery 1	-	≈65-90	-	-
Aft Solar Array (Battery Charge Array 2)		1	1	1	1	1	T = 0 64 L = 26 3 D = 57 6	30 31	Primary power source for bus 2 and battery charge power for battery 2	-	≈65-90	-	-
Battery, 6 Cells		6	6	6	6	6	3 4 x 3 26 x 4 81	4 08	Secondary power source for bus 1 and 2	-	6 A-H	-	-
Battery Discharge Control (Included 4 Cell Battery)		2	2	2	2	2	3 4 x 3 6 x 4 8	3 85	Battery discharge control for each battery	Series regulator	≈86	No	Yes
Bus Current Sensor		4	4	4	4	4	1 84 x 1 94 x 2 81	0 3	Current sensors for bus 1 and 2	Magnetic amplifier	0-3 5A	-	-
Bus Relay		1	1	1	1	1	2 06 x 2 12 x 2 36	0 34	Paralleling buses on command	Relay driver		Yes	Unpara at 21 V
Bus Voltage Limiter		4	6	4	6	6	1 94 x 2 41 x 3 5	0 62	Clamp upper bus limit (2 limiters in ATS A,D, E, are used for thermal control)	Shunt regulator	39	-	-
Current Control Unit			1		1	1	1 15 x 1 98 x 2 23	0 13	Driver for bus voltage limiter	Voltage driver	13 5	Yes	-
Switching Converter						1	3, 5 x 4 02 x 8 78	5 56	Convert high voltage panel to lower voltage at bus	DC-DC switch converter	142	Yes	-
TWT Power Supply	Communication subsystem	4	4	4	4	4	2 79 x 5 79 x 6 82 1 105 (irregular shape)	1 3	Power for TWT tube	Series regulator and dc-dc converter	14 0	Yes	Yes
Command Regulator	Command subsystem	2	2	2	2	2	1 00 x 3 86 x 4 0	0, 5	Power for command system	Series regulator and dc-dc converter	4 15	-	
Telemetry Encoder Regulator	Telemetry subsystem	2	2	2	2	2	3 43 x 5 04 x 0 98	0 5	Power for telemetry system	Series regulator and dc-dc converter	3 4	Yes	
Telemetry Transmitter Regulator (With Transmitter)	Telemetry subsystem	4	4	2	2	4	Estimate 3 43 x 5 04 x 0 98	Est 0 5	Power for telemetry system	Series regulator	4 3		

Table 3-1 (continued)

Unit Name	Used On	Number Required for each Spacecraft					Unit Size, inches	Unit Weight, pounds	Functions	Regulation Technique	Unit Power Rating, watts	Other Provisions	
		B (F-1)	A (F-2)	C (F-3)	D (F-4)	E (F-5)						On-Off	Overload
Repeater Regulator	Communication subsystem	2	2	2	2	2	1.07 x 3.98 x 6.3	0.31	Power for communication system	Series regulator	4.65	Yes	Yes
PACE Regulator	Antenna control	2					1.0 x 3 3 x 6 15	0.5	Power for antenna control	Series regulator and dc-dc converter	6.2		
MACE Regulator	Antenna control			2			1 0 x 3 3 x 6 15	0.5	Power for antenna control	Series regulator and dc-dc converter	6.2		
Subliming Solid Driver	Reaction control		3		3	3	0 9 x 3 42 x 5 08	0.38	Power for reaction control system	Series regulator and static switch	16.8		
Phase Shifter Driver	Antenna control	2					4.08 x 4.66 x 5.08	2.22	Power for antenna control	Shunt regulator and dc-dc converter	5.0		
Payload Regulator for													
Cloud Camera	Experiment	1					1.23 x 3.03 x 3.44	0.31	Power for experiment	Series regulator	14.4		
EME		1				1		0.31			35.0		
Nutation		1						0.30			30.0		
Ion Engine		1			2	2		0.32			18.0		
VHF		2	1	2				0.31			45.6		
MET			2					0.25			40.8		
Albedo			1					0.33			30.0		
Gravity Gradient			1		1	1		0.32			30 0		
Third Harmonic				1		1		0.29			18.0		
Image Dissector Camera				1				0.31			24.0		
Resistojet				1	1	1		0.34			35.0		
Self-Contained Navigation				1				0.30			14.4		
Reflectometer				1				0.30			35.0		
Multicolored Spin Scan				1				0.30			2.4		
Mechanically Despun Antenna				2				0.30			35.0		
Image Orthicon Camera					2			0.24			36 0		
Magnetometer					1	1		0 24			14 4		
Magnetic Damper					1	1		0.25			40 8		
Millimeter Wave						1		0.28			40.8		
Millimeter Wave Backup						1		0.28			24 0		
Solar Cell						1		0.28			14 4		

TABLE 3-2. WEIGHT/VOLUME/PARTS COUNT FOR EXISTING ATS DESIGN

Control Item	Synchronous Altitude Spin Stabilized ATS-B (F-1)		Medium Altitude Gravity Gradient ATS-A (F-2)		Synchronous Altitude Spin Stabilized ATS-C (F-3)		Synchronous Altitude Gravity Gradient ATS-D (F-4)		Synchronous Altitude Gravity Gradient ATS-E (F-5)	
	Weight/ Volume, lb/in.	Parts Count	Weight/ Volume, lb/in.	Parts Count	Weight/ Volume, lb/in.	Parts Count	Weight/ Volume, lb/in.	Parts Count	Weight/ Volume, lb/in.	Parts Count
Power Subsystem										
Battery Discharge Control	2 2/51.8	152	2.2/51.8	152	2.2/51.8	152	2 2/51.8	152	2.2/51.8	152
Current Sensor	1.20/40	108	1.2/40	108	1 2/40	108	1.2/40	108	1.3/40	108
Bus Relay	0 34/10.3	45	0.34/10.3	45	0.34/10.3	45	0.34/10 3	45	0.34/10 3	45
Voltage Limiter	2.48/65 5	100	3.72/98 5	150	2 48/65.5	100	3 72/98.5	150	3 72/98.5	150
Current Control Unit	-		0.13/5.08	34	-		0 13/5 08	34	0.13/5.08	34
Switching Converter	-								5.56/123 5	205
Spacecraft Load Regulator										
TWT Power Supply	5 2/136 5	440	5 2/136.5	440	5 2/136.5	440	5.2/136 5	440	5.2/136.5	440
Command	1.0/30 8	112	1.0/30 8	112	1.0/30.8	112	1 0/30.8	112	1 0/30.8	112
Telemetry Encoder	1.0/33 8	196	1.0/33 8	196	1.0/33.8	196	1 0/33.8	196	1.0/33.8	196
Telemetry Transmitter	1.2/51.2	176	1 2/51.2	176	0.6/25.6	88	0.6/25.6	88	1.2/51.2	176
Repeater	0 62/53.6	244	0 62/53.6	244	0.62/53.6	244	0.62/53 6	244	0 62/53.6	244
PACE or MACE	1 0/40.6	180	-		1.0/40 6	180	-		-	
Subliming Solid Driver	-		0.912/37.2	246			0.912/37 2	246	0.912/37.2	246
Phase Shifter Driver	4 44/193 6	364	-		-		-		-	
Experiment Payload Regulator	1 86/76 8	344	1.46/64	275	3 06/128	620	2.27/102	496	3.23/141	682
TOTAL	22.54/783 9	2461	18 98/612.78	2178	18.7/616.5	2285	19 19/625.18	2311	26 31/813.28	2790

TABLE 3-3. EFFICIENCY CHART FOR EXISTING ATS DESIGN

	Synchronous Altitude Spin Stabilized ATS-B (F-1)				Medium Altitude Gravity Gradient ATS-A (F-2)			
	Beginning of Life		End of Life		Beginning of Life		End of Life	
	Output Power, watts	Power Dissipation, watts	Output Power, watts	Power Dissipation, watts	Output Power, watts	Power Dissipation, watts	Output Power, watts	Power Dissipation, watts
Power Subsystem								
Battery Discharge Control		1 20		1 08		1 20		1.08
Current Sensors		1 22		1 11		1 22		1.11
Harness Drops		1 30		1.10		0.90		0 81
Typical Spacecraft Load								
TWT Power Supply (1)	14.0	6 10	14.0	5 0	14 0	6 5	14 0	5 0
Command Regulator (2)	8.3	3.76	8 3	2 68	8 3	4 08	8 3	2.68
Telemetry Encoder Regulator (2)	6 8	2.48	6 8	1 82	3 4	1.35	3.4	0.91
Telemetry Transmitter Regulator (2)	8.6	1 49	8.6	0.97	4 3	0.85	4.3	0.49
Repeater (2)	9 3	1.80	4 65	0.71	4.65	1.11	4.65	0 71
PACE Regulator (2)	12 4	4.64	6 2	1.87				
Phase Shifter Driver (1)	5.0 (Est)	2 2 (Est)	5.0	2 20				
Subliming Solid Driver								
Payload 1	30.0	5 2	30.0	3 45	14.4	3 68	14 4	2.48
Payload 2	35 0	6 8	35 0	3 75	22.0	4.8	14.4	2 48
Total Typical Load	129 4	38.19	118 55	25 74	71.05	25 67	63 45	17 75
Total Typical Load Current	5 4 A				2 96 A		2.64	
Total Power Required	129 4 + 38 19 = 167 6		118 55 + 25 74 = 144 29		71 1 + 26.7 = 96 79		63.45 + 17.8 = 81.25	
Power Available	180w at 26 9V				124.7w at 27.5V			
Efficiency	77 2%		82 2%		73 45%		78.1%	

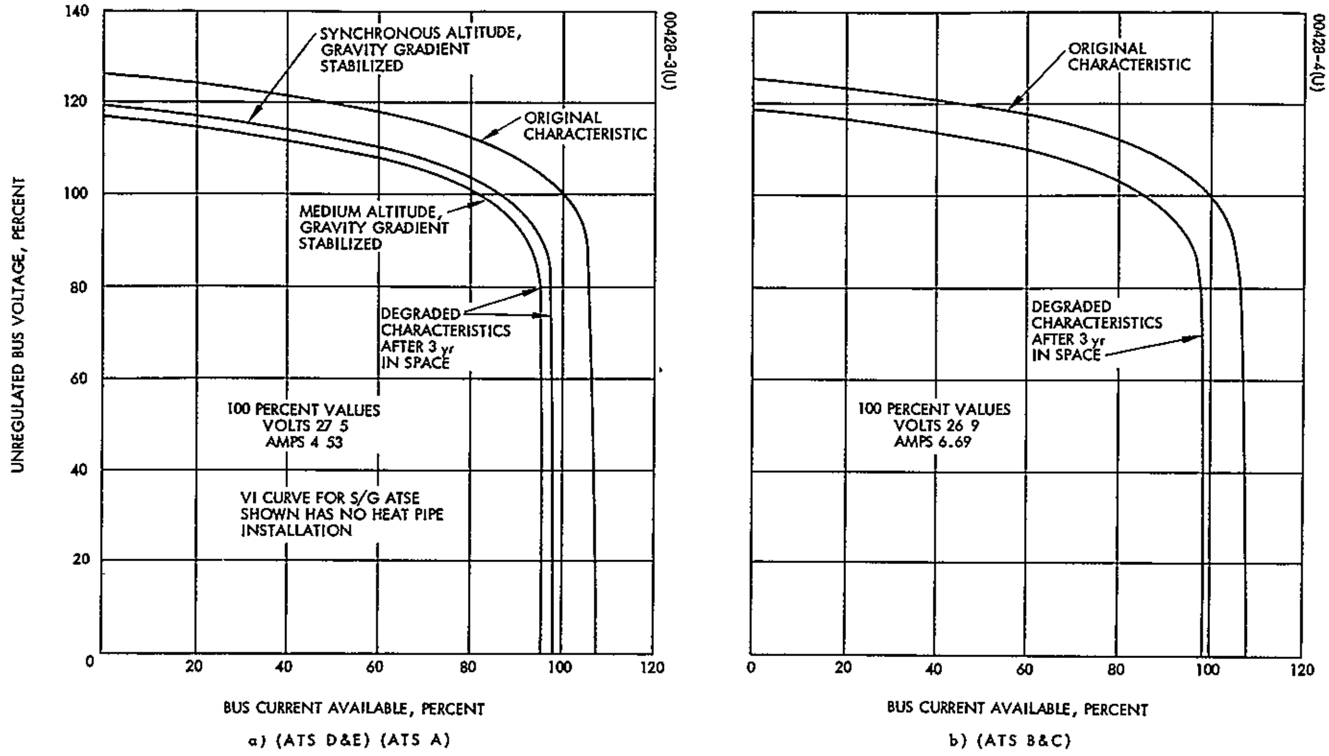


Figure 3-3. Main Solar Array Electrical Power System Characteristics

The remainder of this section describes the present ATS designs in detail.

ATS POWER SUBSYSTEM DESIGN

The ATS power systems consist of forward and aft (main) solar arrays, battery charge arrays, batteries, battery discharge controls, current sensors, a bus relay, bus voltage limiters, a current control, and a switching converter. A description and performance characteristics of the power subsystem units follows.

Solar Arrays

There are two basic solar array configurations. In the one used in the medium altitude gravity gradient (M/G) (ATS-A) and synchronous altitude gravity gradient (S/G) (ATS-D and -E), the two solar arrays are mounted at either end of the spacecraft on the outer cylindrical shell. The forward array has 46 parallel connected subgroups of n on p silicon solar cells. Each subgroup is composed of three parallel strings of 80 cells in series. The series cells are divided into two sections, one of 54 cells extending the full length of the array, and the other of 26 cells sharing the length with the adjacent subgroup. The aft array has 44 parallel subgroups. The remaining space is used for the parallel high conductance silicon diodes which connect the subgroups to the unregulated bus. The pertinent solar cell characteristics for this configuration are defined in Table 3-4. The main solar array characteristics are defined in Table 3-5. The power output characteristic curve of the parallel arrays is shown in Figure 3-3a.

The solar array configuration used in the synchronous altitude spin stabilized (S/S) (ATS-B and -C) is two solar arrays comprising the outer cylindrical shell of the spacecraft. The aft segmented array contains a total of 196 strings arranged mainly in parallel groups of three, but also in single and double strings where necessary in order to utilize all the panel area. Each group is made up of 62 cells in series running the full length of the panel. The forward segmented array has a total of 182 strings. As in the case of M/G and S/G, each subgroup of solar cells is connected to the unregulated bus through a pair of high conductance silicon diodes. The main solar array characteristics are defined in Table 3-6. The power output characteristic curve of the paralleled arrays is shown in Figure 3-3b.

Power measurements for the cells and arrays defined in Tables 3-4, 3-5, and 3-6 are based on the following conditions:

Air mass zero

Temperature: 25° C

Solar constant: 140 mw/cm^2

Incidence angle: 90 degrees

Coverslide installed

TABLE 3-4. SOLAR CELL CHARACTERISTICS

Type	N on p silicon
Size	1 x 2 cm
Output power at 0.445 volt	26 mw, average
Output current at 0.445 volt	58.5 ma, average
Nominal bulk resistivity	10 ohm-cm
Coverslide material	Corning 7940 fused silica
Coverslide thickness	0.030 (30 mils)

TABLE 3-5. MAIN ARRAY CHARACTERISTICS - M/G (ATS-A) AND S/G (ATS-D AND -E)

	ATS-A	ATS-E
Maximum power, watts	125	150 (130)*
Maximum current, amperes	4.53	4.44A (4.73)*
Rated voltage, volts	27.5	33.75 (27.5)*
Total solar panel length, inches	47	47
Total number of solar cells (main arrays)	21,600	21,600
Peak panel temperature, °F	150	60
Number of cells in series string	80	80
Number of strings	270	270
Boom shadowing loss, estimated percent	5	5

*Bracketed figures show values if heat pipe does not function or if switching converter is bypassed.

TABLE 3-6. MAIN ARRAY CHARACTERISTICS - S/S (ATS-B AND -C)

Maximum power	180 watts
Maximum current	6.69 amperes
Rated voltage	26.9 volts
Total solar panel length	52.5 inches
Total number of solar cells (main arrays)	23,436
Number of cells in series string	62
Number of strings	378
Shadowing	0

Battery Charge Arrays

The battery charge array for M/G and S/G has 11 cells in series; the array for S/S has 15 cells in series. M/G has 24 strings in parallel, and S/G and S/S have 12 cells in parallel. The series strings are symmetrically spaced in strips of one cell width at 60 degree intervals around each solar array cylinder. Thus, for M/G there are four parallel strings at each location and for S/G and S/S two parallel strings. The current capability for each array at battery charge voltages is 0.40 ampere for the M/G arrays and 0.20 ampere and 0.21 ampere for the S/G and S/S arrays respectively. The battery charge array compositions are shown in Table 3-7.

TABLE 3-7. BATTERY CHARGE ARRAYS

	M/G	S/G	S/S
Series solar cells for charge	11	11	15
Parallel solar cells for charge/battery	24	12	12
Total solar cells required for charge/battery	264	132	180
Current capacity at battery charge volts	0.40	0.20	0.21

Battery

Each unregulated spacecraft bus is powered during transient and eclipse load operation by one 22 cell nickel-cadmium battery. The cell characteristics are listed in Table 3-8. A self-contained battery discharge control allows battery discharge only when the bus approaches -24.5 volts, whereupon the battery starts to assume the load (as the eclipse commences).

M/G. The M/G orbital period is 6.4 hours, and the maximum shadow time is 0.75 hour. The charge array configuration limits the charging current to 0.4 ampere so that with the expected battery efficiency the discharge current during the maximum eclipse periods may be permitted to be as great as 1.75 amperes and still have the battery fully recharged each orbit. The maximum depth of discharge is therefore about 22 percent. The maximum possible number of discharge cycles per year for the given orbit is 1370. At the expected temperature environment (40 to 100°F) and the depth of discharge given above, the battery has an expected life in excess of 10,000 cycles.

S/G and S/S. The S/G and S/S charge arrays have a current capability of 0.2 ampere. The orbital period is 24 hours, and the maximum shadow time is 1.2 hours. Thus, the maximum discharge current capability during the longest eclipse period (in order to allow for full recharge on each orbit) is 2.35 amperes per battery. This results in a battery depth of discharge of 45 percent for the longest duration eclipse. The semiannual eclipse season at synchronous altitude is only 45 days; therefore, the cycle life requirements over the 3 year expected life are only 270 cycles.

Battery Discharge Control

The discharge regime of the spacecraft batteries is controlled by integral battery discharge controls. These controls allow the batteries to discharge only when their associated spacecraft buses approach -24.5 volts.

TABLE 3-8. BATTERY CELL CHARACTERISTICS

Cell type	Nickel-cadmium hermetically sealed
Cell capacity	6 amp-hr
Maximum continuous overcharge capability	500 ma
Nominal discharge voltage	1.2 volts
Cells per battery	22
Battery charge voltage	31.9 volts

The use of this discharge control allows for complete utilization of the maximum power available from the solar array without discharging the spacecraft batteries. The spacecraft batteries will furnish only the additional current required that the main array cannot furnish. To eliminate the possibility that one battery may supply a large portion of the spacecraft load when the buses are paralleled, a current sharing circuit limits current division to a ratio of 40 to 60 percent or better.

Each battery discharge control is a series electronic regulator to provide regulated -24.5 volts from the battery when the solar panel drops below that point. It consists of a reference, differential amplifier, driver, and power transistor in series with the battery packs. The detailed schematic is presented in Appendix A and the basic operation of the control is as follows. The bus voltage is sensed and compared with the reference. The resultant voltage is amplified and determines the power transistor impedance required for regulation. When the buses are paralleled, a set of contacts on the bus relay is utilized to crossconnect the input of the driver amplifiers. This interconnection forces both discharge controls to follow the highest reference, and the drive current divides between two discharge controls according to the series resistor, transistor base-emitter voltages, and battery voltages. This action with gain selected power transistors forces the battery discharge currents to divide with a maximum unbalance of 40 to 60 percent. Performance characteristics for the battery discharge control are presented in Table 3-9.

TABLE 3-9. BATTERY DISCHARGE CONTROL PERFORMANCE CHARACTERISTICS

Set point (with battery voltage -27.0 to -33.5 volts)	-24.6 volt to -24.89 volt dc
Current equalization (between two batteries)	53.6/46.4 percent unbalance
Current	
Steady state	0 to 3.5 amperes
Short circuit - paralleled buses	13.0 to 20. amperes
Short circuit - unparalleled buses	3.5 to 6.0 amperes
Power dissipation	
Control ON	9.3 watts
Control OFF	0.6 watt
Size and weight (including 4 battery cells)	3.4 x 3.6 x 4.8 inches 3.85 pounds
Estimated battery discharge control unit size and weight	1.5 x 3.6 x 4.8 inches 1.1 pound
Reliability (t = 3 years)	0.97735
Schematic diagram	Figure A-1

Current Sensors

Current sensors are used to provide general housekeeping data for the performance of the power system aboard the spacecraft. The power of each solar array bus is carried by two separate wires. Two sensors are used per bus to determine the total solar array current. The outputs of the two sensors are added together in the spacecraft prior to entering the encoder. In addition, sensors located internal to the battery controller are used for battery charging and battery discharging information. Separate sensors have been used to determine the battery currents in lieu of a single biased sensor in order to provide more accurate information on battery charging. This has been done since the battery charge and discharge requirements vary from one another by an order of magnitude. The current sensors are of the magnetic amplifier type (see Appendix A). Table 3-10 lists the current sensor characteristics.

TABLE 3-10. CURRENT SENSOR PERFORMANCE CHARACTERISTICS

Current range	
Bus and battery discharge	0 to 3.5 amperes
Battery charge	0 to 0.5 ampere
Telemetry analog signal voltage	-0.15 to -5.0 volt dc
Power dissipation	0.33 watts
Output ripple voltage	115 mv
Oscillator frequency	26.3 to 27.7 kHz at -28 volts dc
Output impedance	1300 ohms
Size	
Bus current sensor	1.84 x 1.94 x 2.81 inches
Battery charge/discharge sensor	Included in battery discharge control
Weight	0.3 pound
Reliability (t = 3 years)	0.99465
Schematic diagram	Figure A-2

Bus Relay

A relay is included in the spacecraft to provide greater flexibility of payload and electronic subsystem operation. The potential combinations of payload operation could be asymmetric and, without the capability of tying the buses together electrically, operational restrictions would occur. The bus relay is maintained in an open position during launch. After injection into orbit, the relay can be closed and latched upon ground command. However, should a bus fault develop which drops the bus voltage sufficiently, the relay will automatically open and will not reclose except by ground command.

The unit consists of a two position magnetic latching relay, driver circuits, and automatic unparalleling circuit. The schematic diagram for the unit is shown in Appendix A and its operation is as follows. A command pulse on the driver circuit A energizes the relay coil A and connect both buses together. A command pulse on the driver circuit B energizes the relay coil B and disconnects the buses. The circuit using transistor Q3 and zener diode CR11 senses the bus voltage and activates the relay coil B when the bus voltage becomes less negative than -20.65 volts. This again disconnects the buses and protects against a short in the spacecraft. The performance characteristics of the unit are listed in Table 3-11.

TABLE 3-11. BUS RELAY PERFORMANCE CHARACTERISTICS

Power consumption	
Pulse command applied	65 ma at -33 volts for 50 ms
No command signal	0 ma
Automatic unparallel (undervoltage conditions)	-20.65 volts ±0.35 volts
Reset	-22.85 volts ±0.68 volt
Operating time at short circuit	62.5 ±22.5 ms at -24.5 volts 140 ±20 ms at -33.5 volts
Size	2.06 x 2.12 x 2.36 inches
Weight	0.34 pound
Reliability (t = 3 years)	0.98936
Schematic diagram	Figure A-3

Bus Voltage Limiter

The function of the bus voltage limiter is to limit the maximum voltage the unregulated bus will see during any transient period such as: 1) emergence from the solar eclipse; 2) S/G bus operation during its period of operation as a spinning satellite in the transfer ellipse when the solar panels are operating at approximately room temperature; 3) operation when the spacecraft load is light. The unit is also used to provide thermal input to the spacecraft in key locations. The schematic diagram of the unit is shown in Appendix A. Each bus voltage limiter section consists of voltage sensing with a reference, driver, and power transistor which shunts current through the resistors from the bus to ground. The second section, packaged with the first section, is identical and the performance characteristics of the unit are shown in Table 3-12.

TABLE 3-12. BUS VOLTAGE LIMITER PERFORMANCE CHARACTERISTICS

Input current	<400 μa at 0 to -31.5 volts 1.2 to 1.45 amperes at 32.2 volts
Bus voltage limit	-31.61 to 32.03 volts
Power dissipation	
Unit OFF	0.0126 watts maximum
Unit ON	46.7 watts maximum
ON-OFF control	By current control unit (for M/G and S/G only)
Size	1.94 x 2.41 x 3.5 inches
Weight	0.62 pound
Reliability (t = 3 years)	0.96615
Schematic diagram	Figure A-4

Current Control Unit

The current control unit is used in the ATS-A, -D and -E (M/G and S/G) spacecraft. The function of the unit is to shunt a fixed current into appropriate bus voltage limiters on command, thus providing the spacecraft with a source of heat. The unit is designed to shunt a total of 1 ampere into the bus voltage limiters when commanded. The schematic diagram of the unit is shown in Appendix A. The circuit works as follows.

The latch switch changes states by ground command. A positive 15 volt pulse applied at CR5 or CR6 turns on Q2. This transistor turns on Q1. During the ON state, all transistors are in saturation. For the ON state of the latch switch, zener diodes CR15 and CR16 apply a constant voltage across the base to emitters of the power transistors and the 22 ohm resistor in each voltage limiter. This causes approximately 250 ma dc to be conducted in each voltage limiter. A positive 15 volt pulse applied to CR3 or CR4 turns off the latch switch and removes the constant voltage from each voltage limiter. Performance characteristics of the unit are listed in Table 3-13.

TABLE 3-13. CURRENT CONTROL UNIT PERFORMANCE CHARACTERISTICS

Power dissipation	
Unit ON	0.43 watt
Unit OFF	0 watt
ON-OFF control	
Command voltage	+15 to +20 volts dc (2 ma current sink) for 50 ms from 10 kilohm source impedance
Quiescent (no command)	-24 volts ±3 percent (100 kilohm) source impedance
Size	1.15 x 1.98 x 2.23 inches
Weight	0.13 pound
Schematic diagram	Figure A-5

Switching Converter

In the ATS-E (S/G), the switching converter is added in the power subsystem to convert the high voltage produced by the solar array to a lower bus voltage. This high voltage is the direct result of the addition of heat pipes to the spacecraft which cause the solar array to operate at a lower temperature. The switching converters are connected between the main solar array and the unregulated buses. A relay is provided for bypassing the converter. The relay is actuated via ground command. This would allow power supply operation in case of failure of a switching converter and also yield higher bus voltage in case of an overheated solar array.

Table 3-14 shows typical values of the input and output voltages and currents for the switching converter. It shows that the greatest total load current out of the switching converter (5.59 amperes) is appreciably greater than the total input current (4.50 amperes).

TABLE 3-14. SWITCHING CONVERTER PERFORMANCE CHARACTERISTICS

Main solar array voltage	-33.05 volts	-37.5 volts	-41.8 volts
Total solar array current	4.5 amp	3.4 amp	1.5 amp
Switching converter			
Output voltages	-24.5 volts	28.0 volts	31.0 volts
Total output current	5.59 amp	4.22 amp	1.43 amp
Stepdown factor	0.74		
Efficiency			
0.55 ampere load		88.6% at room temperature 87.6% at +150°F 90.1% at -15°F	
2.85 ampere load		93.6% at room temperature 94.3% at +154°F 92.0% at -15°F	
AC ripple			
0.55 ampere load (5 kHz)	0.038 to 0.200 volts (-15° to +150°F)		
2.85 ampere load (5 kHz)	0.065 to 0.200 volts (-15° to +150°F)		
Size	3.5 x 4.02 x 8.78 inches		
Weight	5.56 pounds		
Schematic diagram	Figure A-6		

The schematic diagram of the unit is shown in Appendix A. Operation of the unit is as follows. Z1 is a linear microcircuit amplifier which is used as a free-running multivibrator. Supply voltages for Z1 are developed across the two zener diodes. The 10 kilohm and 100 ohm resistors connected to the non-inverting input provide positive feedback from output to input. The multivibrator ON and OFF time intervals are determined by resistors R2 through R4 in series with the CR1 diodes. The repetition rate of 5 kHz may be varied by changing the C1 capacitor.

The pulse output of the multivibrator drives Q1 buffer stage, Q2 inverting amplifier, and the Darlington-connected output stages that handle the load current. The operating principal of the unit is an oscillating switch connecting the input L-C filter to the output L-C filter. This arrangement permits an efficient transformation of the input power to a lower voltage and higher current at the output. The commutating diode provides a path for current through the output inductor when the pass transistor stage is in the nonconducting state. The performance characteristics of the unit are shown in Table 3-14.

ATS SPACECRAFT LOAD REGULATOR DESIGN

All ATS regulators utilize the basic series regulator circuit and standard driver circuit. The basic circuits for the regulator and the driver used in the ATS design including the payload regulator are shown in Figure A-7. The component values of the basic regulator circuit are selected as required for the particular application. The driver circuit is used to control the regulator circuit. Prior to command activation, both of the driver circuit transistors are turned OFF. Upon application of a positive command pulse, the input transistor conducts, which causes the base of the output transistor to turn ON. The output transistor then functions as a switch which routes the unregulated voltage to the regulator control input for the duration of the command pulse. Application of the unregulated voltage from the solar panel to the regulator control input causes the regulator to turn OFF or to turn ON as applicable.

A description of the ATS spacecraft load regulators is as follows.

TWT Power Supplies

This unit contains the TWT filament inverter, high voltage converter, and ferrite switch drive; in addition to the -24 volt regulator. The FIL ON command turns on the regulator, applying -24 volts to the self-starting filament inverter. Upon turn-on, the -24 volts are also applied to the ferrite switch coil through a saturated transistor switch. A capacitor in the base of this transistor causes the switch to turn off in 60 ± 40 ms. The high voltage converter is not self-starting and requires a separate command to turn on. When on, the high voltage converter supplies nominally 400 volts to the anode, -800 volts to the to the collector, and -1400 volts to the cathode of the TWT. However, the high voltage transformer is wound to give the exact voltages required for a particular TWT. High voltage ripple

limits are 0.5 volt p-p on the anode and 2 volts p-p on the cathode and collector. The filament output is 4.5 volt rms 5 kHz square wave, with current limited to 450 ma into a cold filament. In addition, the regulator supplies -24 volts to the antenna ferrite switch driver.

Detailed performance characteristics for the unit are given in Table 3-15.

TABLE 3-15. TWT POWER SUPPLY PERFORMANCE CHARACTERISTICS

Input voltage			-24.5 to -32.5 volts	
Output voltages	4.5 volts rms	+400 volts	-800 volts	-1400 volts match to TWT
Frequency	5 kHz	-	-	-
AC ripple	-	0.5 volt p-p	2.0 volts p-p	2.0 volts p-p
Output power			14.0 watts	
Power dissipation			6.0 watts at normal input	
Regulation			±1.5 percent typical	
On/off control			Required	
Overload protection				
Regulator			900 ma	
Filament			450 ma maximum	
Size			(2.79 x 5.79 x 6.82) 1.105 inches thick, irregular shape	
Weight			1.3 pounds	
Reliability			*	
Schematic diagram			Figure A-8	

*TWT power supply remains at the TWT and the analysis was not performed.

Command Regulator

The self-starting command regulator unit is composed of a basic regulator circuit and a dc to dc converter which provides +24 volts, -5.5 volts, and -24 volts. The outputs will track the input until the input becomes more negative than -24.50 volts, at which point regulation will commence. This unit contains no on/off circuit since the outputs are required at all times. Two command regulators are required per spacecraft. Each command regulator provides power to a command receiver, command decoder, and spacecraft clock. The performance characteristics of the unit are shown in Table 3-16.

TABLE 3-16. COMMAND REGULATOR PERFORMANCE CHARACTERISTICS

Input voltage	-20 to -34 volts		
Output voltages	-24 volts	+24 volts	-5.5 volts
Load currents	52 ma	106 ma	65 ma
Regulation	±1.4%	±1.08%	±8.0%
Overload protections	130 ma	130 ma	
On/off control	Not required		
AC ripple	50 mi p-p typical		
Output power	4.15 watts		
Power dissipation	1.88 watts at nominal input		
Size	1.00 x 3.86 x 4.0 inches		
Weight	0.5 pound		
Reliability (t = 3 years)	0.99044		
Schematic diagram	Figure A-9		

Telemetry Encoder Regulator

The telemetry encoder regulator unit is composed of four standard driver circuits, two basic regulator circuits, and a dc to dc converter. The unit accomplishes two essentially separate functions upon commands received from the command decoder. One function provides a regulated -24 volt output and the other provides -24, +22.5, +14, and -14 volt outputs. The performance characteristics are shown in Table 3-17.

TABLE 3-17. TELEMETRY ENCODER REGULATOR PERFORMANCE CHARACTERISTICS

Input voltage	-24.5 to -32.5 volts			
Output voltages				
PCM mode	-24 volts	+14 volts	-14 volts	+22.5 volts
Load current	62 ma	25 ma	49 ma	24 ma
Regulation	±0.1%	±1%	±1%	±1%
Overload protection	150 ma			
SCO mode	-24 volts			
Load current	15 ma			
Regulation	±0.1%			
Overload protection	30 ma			
Output power	3.4 watts			
Power dissipation	1.24 watts at nominal input			
On/off control				
PCM	Required			
SCO	Required			
AC ripple	50 mv p-p typical			
Size	3.43 x 5.04 x 0.98 inches			
Weight	0.5 pound			
Reliability (t = 3 years)	0.98707			
Schematic diagram	Figure A-10			

Telemetry Transmitter Regulator

The telemetry transmitter regulator is composed of two standard driver circuits and a basic regulator circuit which provides a -24 volt output to the telemetry transmitter. Each telemetry transmitter requires an

individual regulator unit since the regulator circuits provide on/off control of the respective transmitter, and this regulator is physically packaged with the transmitter. Each regulator exhibits the performance characteristics shown in Table 3-18.

TABLE 3-18. TELEMETRY TRANSMITTER REGULATOR PERFORMANCE CHARACTERISTICS

Input voltage	-24.5 to -32.5 volts
Output voltage	-24 volts
Load current	200 ma
Regulation	±1 percent
AC ripple	3.5 mv p-p typical
Overload protection	240 ma
Output power	4.3 watts
Power dissipations	0.745 watt at nominal input
On/off control	Required
Estimated size	1.23 x 3.03 x 3.44 inches
Estimated weight	0.3 pound
Reliability (t = 3 years)	0.99333
Schematic diagram	Figure A-11

Repeater Regulator

The repeater regulator is composed of four standard driver circuits, two logic matrices, and three basic series regulator circuits (only one on at a time). The unit provides power to the active circuits of the integrated communication repeater for multiple access (M), frequency translation (F), or camera (C). Mode of operations is in response to command inputs. Regulator voltage sensing is at the MFC output. The regulator selected will maintain -23 volts ± 1 percent at this point from 0 to 200 ma load and normal line and temperature variations. Other outputs will track the MFC output, but with wider tolerances. The MF and FC outputs will still have a nominal -23 volt output. The M, F, and C outputs will be one diode drop more negative, or nominally -23.7 volts. The performance characteristics are shown in Table 3-19.

TABLE 3-19. REPEATER REGULATOR PERFORMANCE CHARACTERISTICS

Input voltage	-24.5 to -32.5 volts					
	M	F	C	MF	FC	MFC
Output voltages	-23.7 volts	-23.7 volts	-23.7 volts	-23.0 volts	-23.0 volts	-23.0 volts
Load current	200 ma					
Regulation	±1%	±1%	±1%	±1%	±1%	±1%
Overload	260 ma					
Output power		4.65 watts				
Power dissipation		0.89 watt at nominal input				
On/off control		MFC mode				
		FMC mode				
		CMF mode				
		MFC mode				
Size		1.07 x 3.98 x 6.30 inches				
Weight		0.31 pound				
Reliability (t = 3 years)		0.98021				
Schematic diagram		Figure A-12				

PACE (or MACE) Regulator

The PACE regulator is composed of three standard driver circuits, a basic regulator circuit, and a dc-dc converter which provides a regulated +24 volts dc output. The dc-dc converter is provided with a separate OFF driver circuit to assure that the +24 volt output can be turned off in the event of a malfunction of the basic regulator control circuit. The MACE regulator is identical to the PACE regulator except that the VHF current limiters are not employed. The performance characteristics for both PACE and MACE regulators are given in Table 3-20.

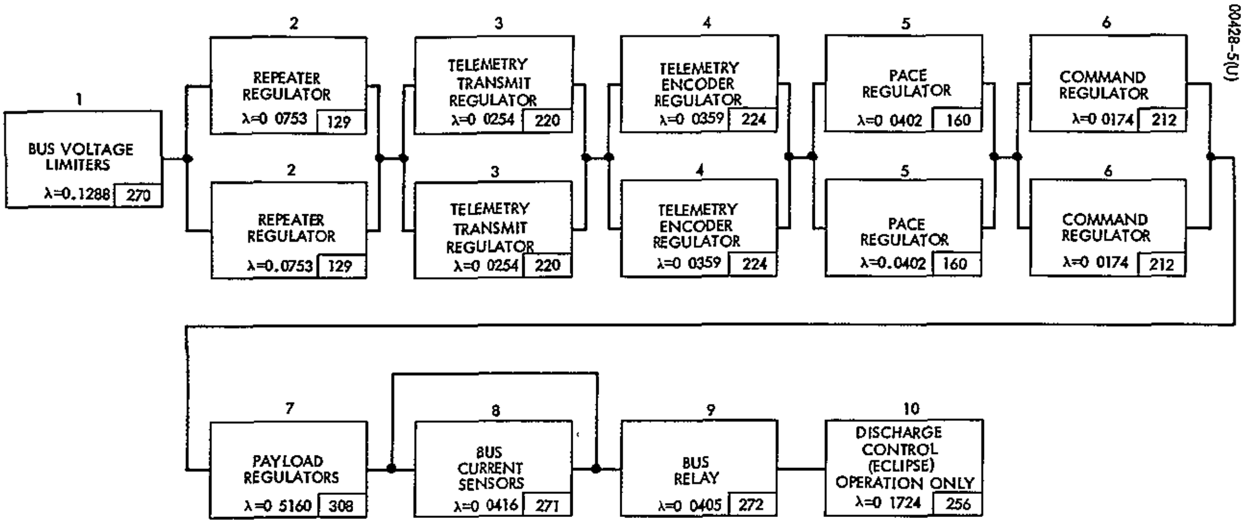
TABLE 3-20. PACE (OR MACE) REGULATOR PERFORMANCE CHARACTERISTICS

Input voltage	-24.5 to -32.5 volts	
Output voltages	+24 volts	-24 volts
Load current	147 ma	114 ma
Regulation	≈±2%	≈±1%
AC ripple	28 mv peak	25 mv p-p
Overload	200 ma	150 ma
Output power	6.20 watts	
Power dissipation	2.32 watt at nominal input	
On/off control	Regulator ON/OFF	
	Converter OFF	
Size	1.00 x 3.3 x 6.15 inches	
Weight	0.5 pound	
Reliability	≈0.98444	
Schematic diagram	Figure A-13	

Subliming Solid Driver

The subliming solid driver is used in ATS-A, -D and -E and contains a -24 volt regulator of 700 ma capacity. When turned on, power is supplied to the subliming solid reaction control system temperature control. The -24 volts are also applied to either the auto-thrust or the override switches. These drivers each have a 700 ma capacity. A digital-to-analog converter indicates the ON/OFF status of the regulator and two drivers. In addition to the line input, 14 volts and -10 volts are required for operation of the digital-to-analog converter. These voltages are obtained from the signal conditioning unit.

The performance characteristics of the unit are similar to those for the ATS experimental payload regulator.



RELIABILITY MATHEMATICAL MODEL (WHERE Q = 1 - R)

$R = (R_1)(1 - Q_2^2)(1 - Q_3^2)(1 - Q_4^2)(1 - Q_5^2)(1 - Q_6^2)(R_7 R_9 R_{10})$

MISSION TIME OF 3 yr GIVES POWER SUBSYSTEM RELIABILITY OF

R = (0 96615) (0 99961) (0 99996) (0.99991) (0.99989) (0 99998) (0 93220) (0 98936) (0 97735)

= 0 8703

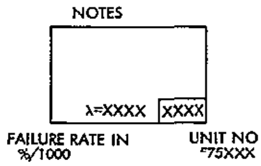


Figure 3-4. Modified ATS Power Subsystem Reliability Logic Diagram

Phase Shifter Driver

The phase shifter driver provides the voltage and power gain for a PACE to drive the ferrite phase shifters upon command from the command decoder. The inputs to the phase shifter driver are the PACE waveform generator output signals. The phase shifter driver contains 16 power amplifiers, 8 dc-dc converters, 8 voltage limiter circuits, and 4 control circuits (two OFF drivers, and two ON drivers). In addition, it contains 16 commutator switches which enable the power amplifier outputs to be telemetered.

The total phase shifter driver is split into two symmetrical units for packaging. These units are designated phase shifter driver 1 and phase shifter driver 2. Thus, each packaged unit contains 8 power amplifiers, 4 dc-dc converters, 4 voltage limiter circuits, 1 ON driver, 1 OFF driver, and 8 commutator switches. This unit is required in ATS-B only.

ATS EXPERIMENT PAYLOAD REGULATOR DESIGN

The spacecraft experiments are powered from the payload regulator. This unit has a 2 ampere capability and is composed of two standard driver circuits, each with dual input command circuits, and a basic regulator circuit modified to provide greater load current handling. The overcurrent limit is set according to the actual full load current of the experiments. Because of the relatively high load current, the overcurrent limit has been made relatively insensitive to line voltage. At -24.5 volts and 40°F, the overload capability is 20 percent, increasing to 70 percent at -32.5 volts and 100°F. Otherwise, this regulator has characteristics similar to other ATS regulators. Typical performance characteristics for the unit are shown in Table 3-21.

RELIABILITY

The ATS has multiple missions, each requiring the successful operation of certain combinations or groups of subsystems or equipment. To compute reliability for every possible mission of each of the three configurations - S/S, S/G, and M/G - would constitute a lengthy task and may not serve the goal of main interest. To obtain a meaningful reliability assessment for the ATS power system only, the existing ATS power subsystem was redefined for this study to include the various subsystem -24 volt series regulators as elements of the power subsystem. ATS reliability predictions were modified to exclude the converters in the command, telemetry encoder and PACE subsystem, TWT power supply, and phase shifter driver unit in the ATS-B (F-1) because it is desired to compare the reliability with a dc centralized regulation system. The reliability logic diagram for the modified configuration is shown in Figure 3-4. Part failure rates used for the reliability predictions are consistent with those used in previous ATS power systems. A mission time of 3 years was used and unit duty cycles are assumed equal to 1.0 except for the payload regulators which have a duty cycle of 0.5.

TABLE 3-21. TYPICAL PAYLOAD REGULATOR PERFORMANCE CHARACTERISTICS

Input voltage	-24.5 to -32.5 volts
Output voltage	-24 volts
Regulation - load	± 0.5 percent
Temperature	± 0.2 percent
Line	± 0.01 percent
Combined effect	± 1.0 percent
AC ripple	1.2 times load current at 40°F
Output power	48 watt maximum
Power dissipation	9.92 watts maximum at nominal input
Size	1.23 x 3.03 x 3.44 inches
Weight	0.34 pound
Reliability (t = 3 years)	0.93220 typical
Schematic diagram	Figure A-14

The results of reliability predictions for the modified ATS power system show that it has a reliability of 0.8703. This reliability assessment is based on no reliability "experience" factors for the part failure rates. If the recommended experience factor of 0.512 is applied, the system reliability changes to approximately 0.9336. A summary of part failure rates, unit reliability predictions, and failure rates for the reliability blocks in Figure 3-4 is shown in Appendix B.

4. ALTERNATE DECENTRALIZED DESIGN

This section presents a brief discussion of alternate decentralized power system configurations that have been used on other spacecraft and might have been used in place of the existing ATS decentralized system. Only qualitative data is presented along with a summary of advantages and disadvantages. With respect to the overall centralization study, it has been concluded that the present ATS decentralized design is sufficiently optimized to provide good comparative data. This conclusion on optimization includes both system configuration and regulator design. While circuit improvement and simplification are always possible, no specific areas of improvement have been seen that would influence the results of this study.

As discussed in Section 3, the ATS spacecraft utilizes a decentralized power distribution system employing the following main components:

1) Two main solar arrays supplying two unregulated buses

2) Shunt limiters across each bus to limit the maximum voltage, hence power dissipation in the remote regulators

3) Two 22 cell batteries, one per bus

4) Two limited power charge arrays, normally operating in the current limit mode, connected in series with the main solar arrays, for battery charging

5) Two series regulator type discharge controls that automatically connect the battery to the bus when the bus voltage drops below 24.5 volts

6) Remote series regulators to provide a controlled input voltage to the utilizing equipment directly or to dc to dc converters which provide higher and/or lower regulated voltages.

The basic block diagram for one of the two typical buses is shown in Figure 4-1. Alternate configurations considered are described in the following paragraphs.

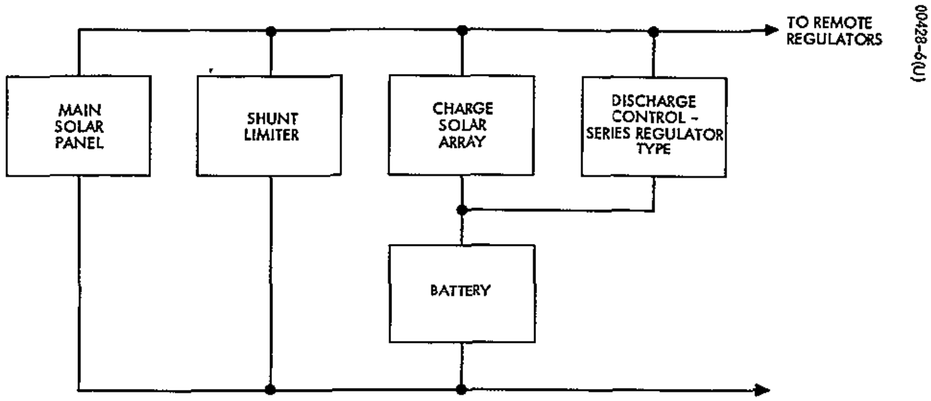


Figure 4-1. Basic ATS Power Distribution System

MODIFIED ATS EMPLOYING SWITCHING REGULATOR TYPE DISCHARGE CONTROL (SYSTEM A)

With the exception of the discharge control, this system (shown in Figure 4-2) is identical to the basic system. By employing the switching regulator in lieu of the series type dissipative regulator, it is possible to realize a higher discharge efficiency. In the series regulator type of discharge control, the delta voltage between the battery terminal voltage and the discharge set point times the load current, which tends to be independent of input voltage because of the remote regulators, represents lost dissipated power. This, therefore, represents a discharge control system with low efficiency at the beginning of discharge when the battery terminal voltage is high, increasing to 90 percent at the end of discharge when the series pass transistor is saturated. The 90 percent figure assumes a saturated drive loss of 10 percent to maintain the pass transistor in saturation. Assuming the same power loss in the switching regulator over the input voltage range as the series regulator in the saturated (end of discharge mode), the actual power saved is as shown in Figure 4-3. This assumption is valid since designing a switching regulator with a 90 percent efficiency at full load is well within the state of the art.

This sytem could introduce a potential EMI problem because of the noise generated by the switching transistor and would require more components than the simple series dissipative regulator, hence would have a lower reliability. Any EMI incompatibility could be corrected by the addition of filters but, again, this would increase the part count and the discharge controller weight. At the ATS power level, the original concept would appear to have a slight edge because the efficiency benefits gained by the switching type discharge would be more than offset by the lower reliability and higher weight. As the power level increases, the power dissipated in the series dissipative regulator would exceed the limits of a single pass transistor with a resultant increase in complexity. The actual crossover

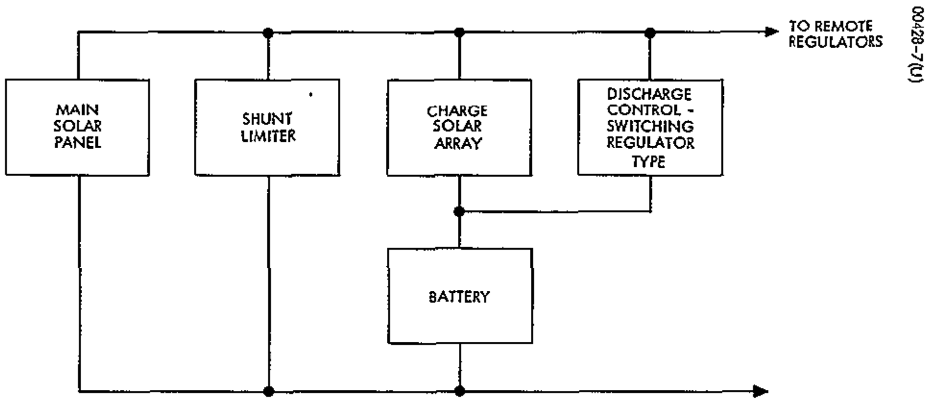


Figure 4-2. Present Decentralized ATS Power System With Switching Regulator Discharge Control

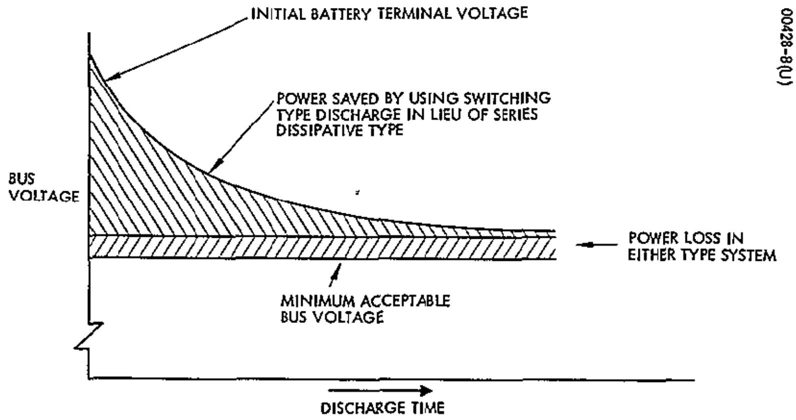


Figure 4-3. Battery Discharge Power Dissipation

power level is dependent upon the anticipated noise susceptability of the utilization equipment and the state of the art of power transistors at the time of decision.

MODIFIED ATS EMPLOYING SWITCHING TYPE DISCHARGE CONTROL AND REMOTE REGULATORS (SYSTEM B)

This system is the same as system A except that switching type remote regulators (STRR) have been considered. By employing STRR, it would be possible to improve the overall system efficiency during both eclipse (as in system A) and non-eclipse operation by permitting the solar

panel bus voltage to operate at its maximum power point. At beginning of life, this would permit an increase in experiment loads. However, with the decentralized system utilizing multiple STRR, there are a number of potential problems. Synchronization of the numerous STRR would have to be considered if EMI was critical and sufficient input filtering would be required on all STRRs to present a relatively constant load on the solar panel. In addition, the STRRs would have to incorporate intelligence in the voltage control system to prevent a possible operating mode that would extract energy from the batteries even when the solar panel could support the load as shown in Figure 4-4.

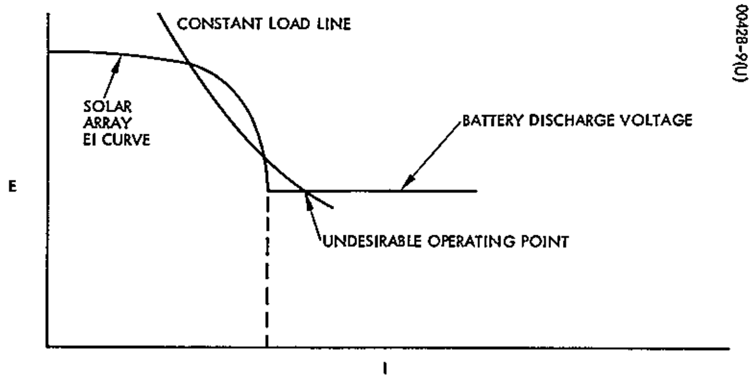


Figure 4-4. Switching Regulator and Solar Panel Characteristics

In summary, this system would provide a higher beginning of life efficiency but would present a number of technical problems - i.e., EMI operating power point, response - that would undoubtedly increase the cost of the design and development phases of a new program and result in lower overall reliability.

RELAY DISCHARGE CONTROL SYSTEM (SYSTEM C)

System C (shown in Figure 4-5) employs a magnetic latching relay to connect the battery to the bus when the bus voltage drops below the 24.5 volt set point. The backup relay provides manual control in the event of an automatic relay failure in either position. Since closure of the relay will raise the bus voltage to the battery terminal voltage less the diode drop, the relay will not automatically return to the OFF state when the load is reduced (assuming a transient pulse load) or the main solar panel is illuminated. If a relatively constant load is anticipated, it would be possible to control the opening of the discharge relay by employing a power or possibly a sun lock on sensor, using appropriate delays. If the electrical system is

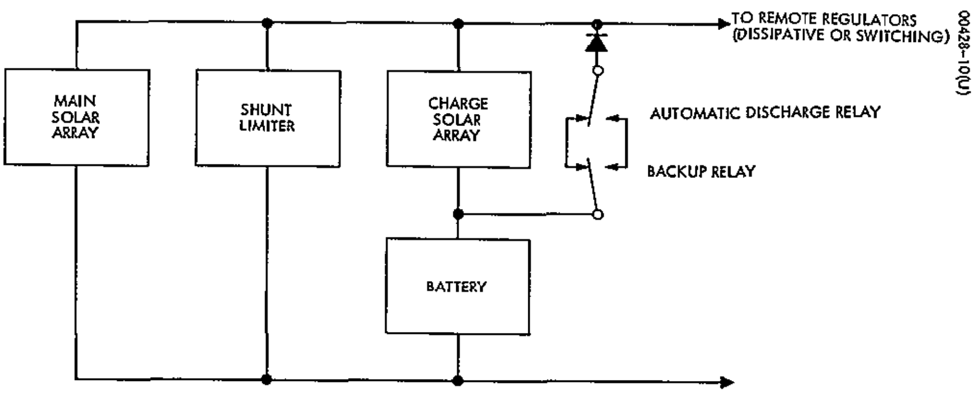


Figure 4-5. Relay Discharge Control System

to be subjected to continuous pulse loads requiring supplemental energy from the battery, this relay discharge concept would not be desirable because of the excessive relay cycling. Also the need for control intelligence to anticipate the need for closure (relay circuit would have a minimum of 5 ms delay) would be required as would the intelligence for reopening the relay circuit.

The main benefit of this system is that it reduces the drop in the discharge control, which could result in a one cell reduction in the battery with associated size, weight, and cost savings. In summary, this approach would warrant serious consideration in a spacecraft where the loads are well defined and constant if a simple ground command system for discharge termination could be employed. If, however, the possibility of varying loads exists, as is the case with the ATS type spacecraft, the control system complexity would influence the system decision in favor of the simpler automatic dissipative discharge control system originally selected.

DC TO DC BOOST CONVERTER CHARGE SYSTEM (SYSTEM D)

In System D (shown in Figure 4-6) the solar array charge string is replaced by a dc to dc regulator. This regulator would have a voltage boost capability high enough to fully charge the battery. It would normally be operating in a current regulated mode. The charge system could be implemented such that the charge current was interrupted when the battery was fully charged by the addition of a temperature biased voltage limit control in the boost regulator or by employing some other means (i. e., signal electrode, coulometer, etc.) for terminating the constant current charge.

The main benefit of this system would be the increase in the main solar panel area, resulting from deletion of the charge arrays. During the normal charge mode, the panel power requirements would be increased due

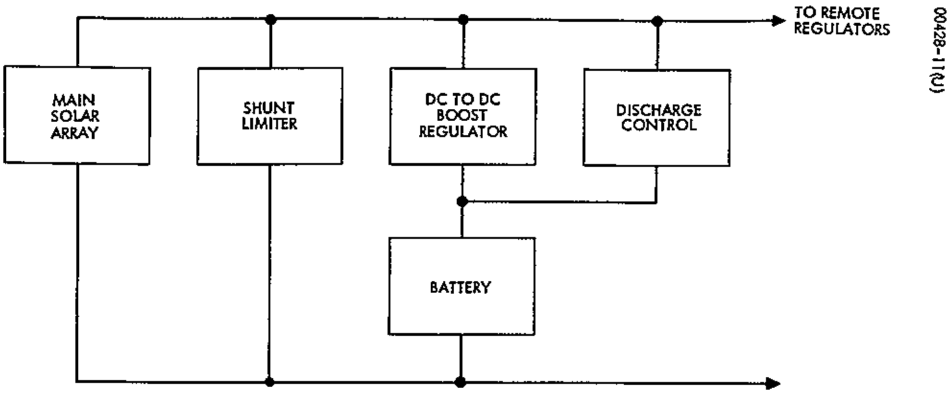


Figure 4-6. DC to DC Boost Converter Charge System

to the loss in the boost circuit. After charge is terminated, however, the total array power will be available for powering spacecraft loads. This will result in a power increase, assuming the present charge array solar cells can be configured to completely utilize their power capability to supplement the present main array.

The main disadvantages of this system would be the added complexity of the dc to dc boost regulator with the resultant reduction in reliability and added power system size and weight. This regulator would also generate EMI that would have to be filtered and/or the system would have to be thoroughly analyzed to ensure system compatibility.

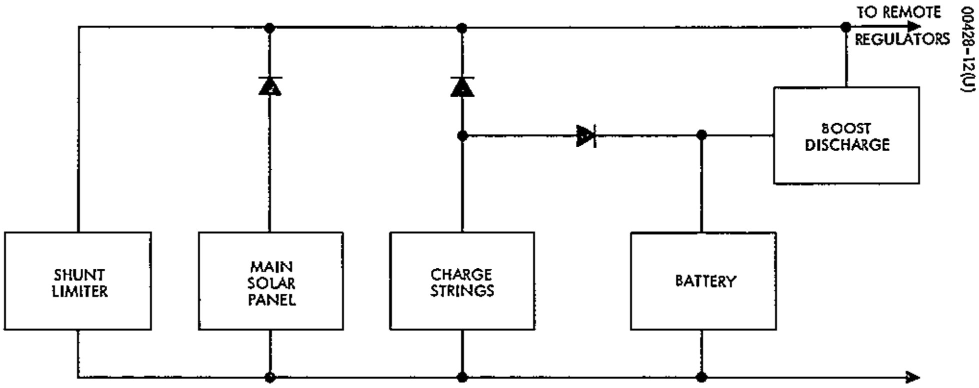


Figure 4-7. Boost Add Discharge System

BOOST ADD DISCHARGE SYSTEM (SYSTEM E)

System E, shown in Figure 4-7, uses a low voltage battery which is then boosted to -24.5 volts by a boost discharge circuit. The boost discharge circuit operates only when the bus drops to its set point. The battery is charged from charge strings which are part of the main array. One out of approximately every 15 or 20 parallel arrays is diode coupled to a charge bus which then charges the battery.

The advantages of this system are that fewer battery cells are needed due to the lower voltage required and the entire panel can be made available to the load. Disadvantages would be similar to those discussed above for other switching types of discharge and charge controls

5. CENTRALIZED DESIGN

This section provides a description of centralized designs meeting the requirements of Section 2. These centralized designs replace all the remote regulators used on the present power system with the exception of the regulators used for the TWT power supply. The TWT regulator-converters are considered an integral design and were not replaced.

CENTRALIZED REGULATOR DESIGN

A number of centralized regulator designs were considered for this application. The one selected was on the basis of minimizing the impact on the present solar panel and battery and requiring the least number of control assemblies. Proceeding in this manner permitted elimination of the bus limiters and discharge control. The system is implemented in accordance with block diagram of Figure 5-1 and distributed to the loads in accordance with Figure 5-2.

Referring to Figure 5-1, the two bus system of the present system is retained. The solar panels are laid out in the same space envelope as before. However, all series strings now have the same number of cells as was previously used for battery charging. For the spin stabilized version, for example, the present design uses 62 cells for the main arrays and 15 additional cells for the charge arrays. The new design uses 77 cells in series for the main arrays. A number of strings (12 for the S/S) are paralleled through isolation diodes and used to charge the 22 cell battery, resulting in the identical charge power as before.

In this configuration, if not charging, all the array power is available to the load. This is not the case in the present design. The 77 series cell array is connected to the load through a switching regulator. The switching regulator provides-24 volts output from the maximum panel voltage to the minimum battery voltage.

Each switching regulator is designed to supply the entire spacecraft load and thus the system is redundant. This redundancy is supplied through paralleling relays on both the input side and output side of the regulators. Paralleling on the input side is the normal function which parallels the solar panels and permits the load on either bus to take as much of the panel power as required up to the total panel (or battery) capacity.

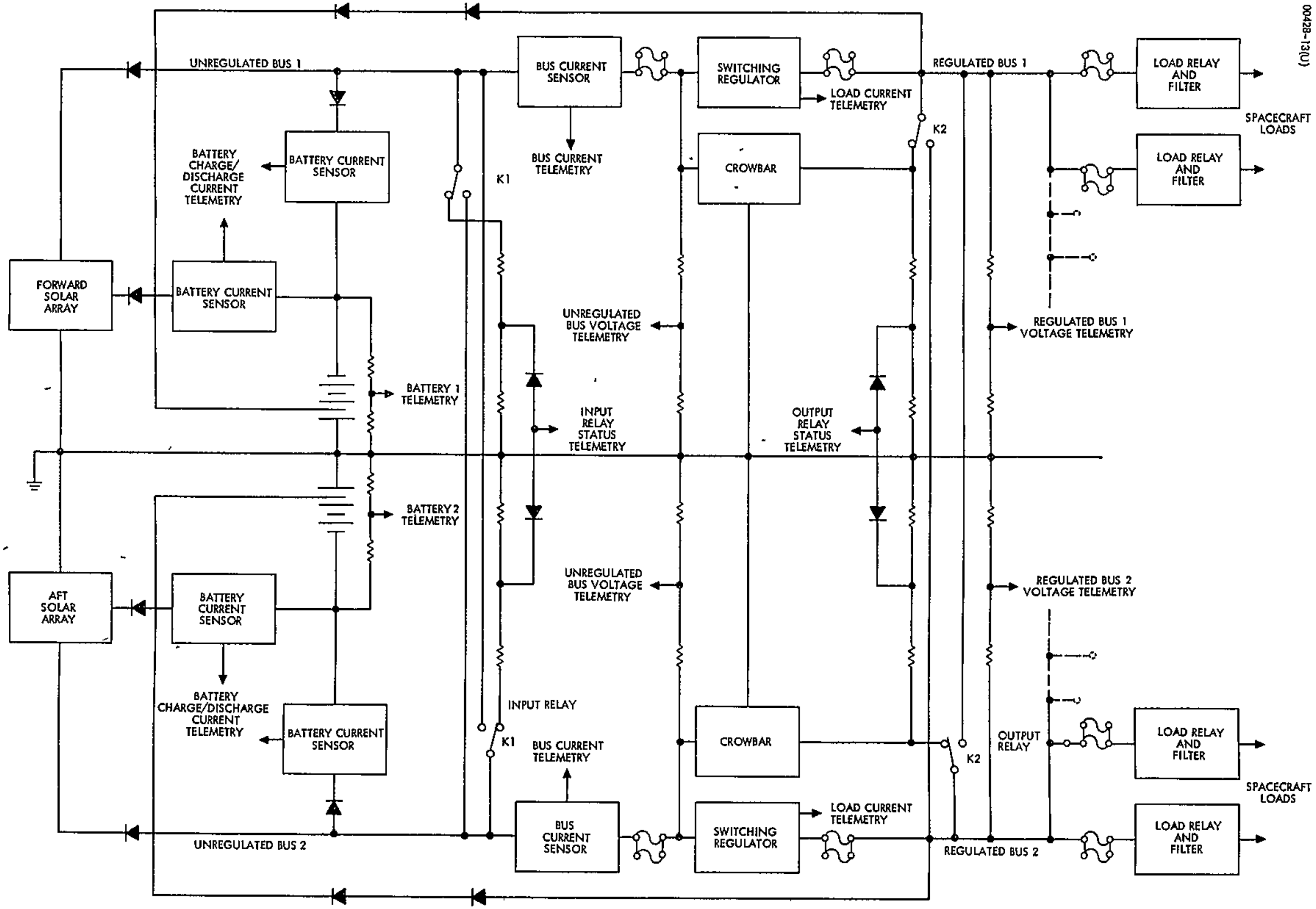


Figure 5-1. Centralized Regulator Design

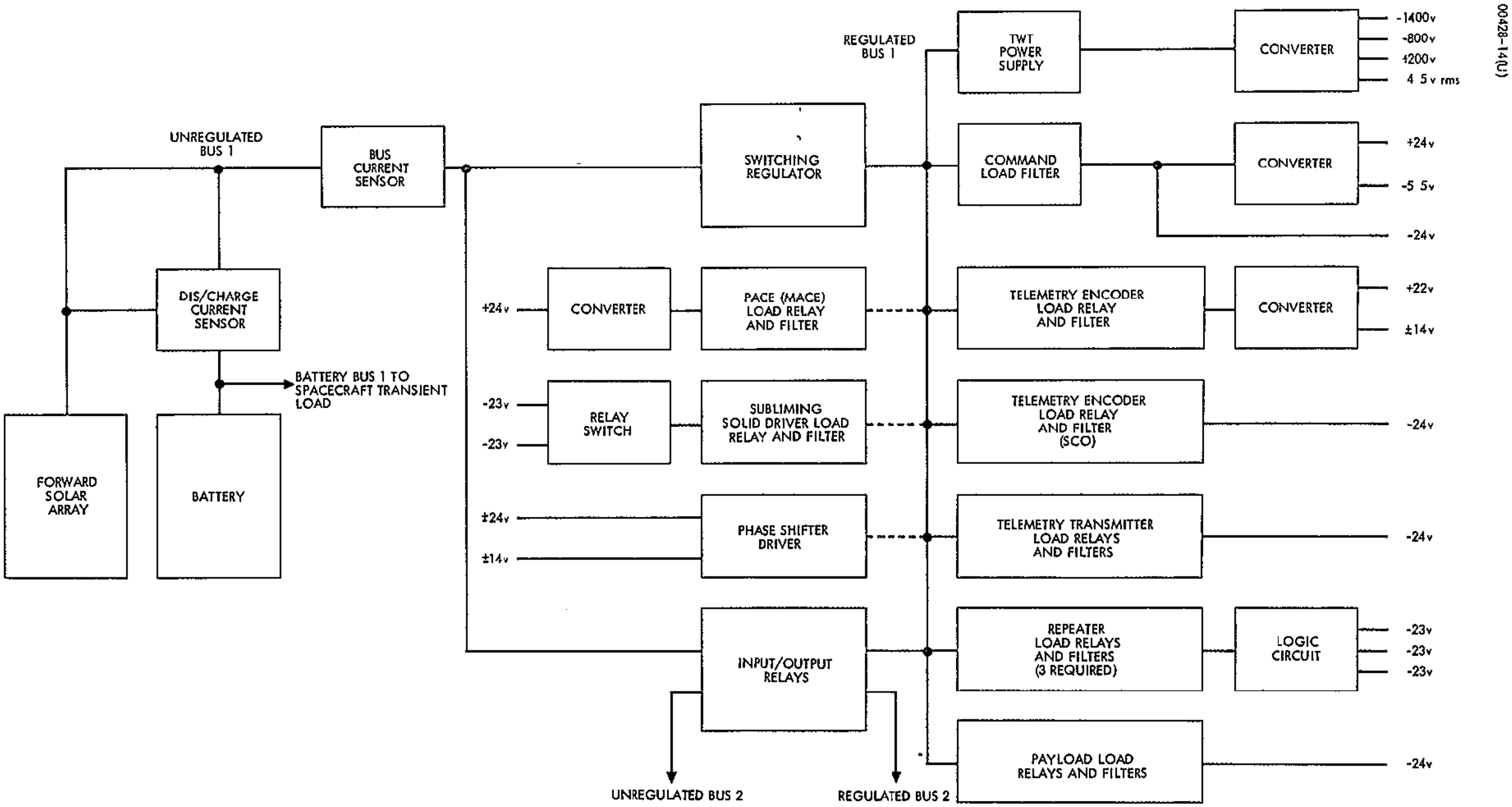


Figure 5-2. Block Diagram of One-Half DC Centralized Regulation Design

The relay on the output side of the regulators is part of the protection feature of the system. Under certain failure modes when one switching regulator is inoperative the output relay is closed, permitting all the connected loads to be powered by the remaining regulator.

Protection against excessive voltage on the bus is provided by a crowbar circuit which senses the output voltage and places a low impedance path across the bus to prevent excessive output. This protection is primarily for a shorted switching regulator which would place the panel right across the bus. When the output relay is paralleled, the crowbar circuit is removed so that both buses will not be disabled. A short on the switching regulator output filter to ground will lower the bus voltage sufficiently to cause a current high enough to blow the output fuse to be drawn through the two series diodes connected to a tap on the battery. This tap is required because the switching regulators are current limited and would not provide enough current for the fuse to flow. Current sensors, voltage, and status signals are as shown.

The regulated bus is connected to the load through commandable relays, fuses and decoupling filters. Figure 5-2 shows half of the total bus system which is designed to be completely equivalent to the present system. This configuration is practical because most of the ATS loads will operate with voltage regulation as provided by a centralized regulator.

Weight, volume, and parts count for this system for all five spacecraft are presented in Table 5-1. Output power, power dissipation, and efficiency at both beginning and end of life for the ATS-B (spinning-synchronous) and the ATS-A (medium altitude gravity stablized) are shown in Table 5-2. Tables showing identical data for the present system were presented in Section 3. A direct comparison can therefore be drawn for the two systems. Additional comparative tables will be shown in Section 6. In assessing the parts count for this system it should be understood that during any development of electronic circuit from paper design to finished hardware parts are usually added in order to improve the system performance. This factor has not been added in this analysis. It is not felt, however, that it would have a marked effect on the results.

The following data describes the centralized regulator design.

Solar Arrays

The solar array configuration for the dc centralized regulation system with respect to the solar cell characteristics, size, weight, and volume is basically identical to the existing ATS designs. The difference is that the solar array now provides the maximum power point at -33 volts for S/S (ATS-B and -C), and -31 volts for S/G and M/G (ATS-A, -D, and -E). The V-I curves for reconfigured solar arrays are shown in Figures 5-3 and 5-4. The solar array for M/G and S/G

TABLE 5-1 WEIGHT/VOLUME/PARTS COUNT CHART FOR DC CENTRALIZED REGULATION DESIGN

	Synchronous Altitude Spin-Stabilized ATS-B (F-1)		Medium Altitude Gravity Gradient ATS-A (F-2)		Synchronous Altitude Spin Stabilized ATS-C (F-3)		Synchronous Altitude Gravity Gradient ATS-D (F-4)		Synchronous Altitude Gravity Gradient ATS-E (F-5)	
	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count
POWER SUBSYSTEM										
Battery Discharge Unit with current sensory and Telemetry	2 08/49	82	2 08/49	82	2 08/49	82	2 08/49	82	2 08/49	82
Input/Output Relay	0.55/16 7	40	0 55/16 7	40	0 55/16 7	40	0 55/16 7	40	0 55/16 7	40
Switching Regulator	6 4/155	230	6 4/155	230	6 4/155	230	6 4/155	230	6 4/155	230
Load Relay and Filter	5 5/167	666	7 0/212	762	6 27/190	726	7 58/230	791	8 83/268	936
Subtotal	14 53/378 7	1018	16 03/432.7	1114	15 3/410 7	1078	16 61/450 7	1143	17 86/488 7	1288
SPACECRAFT LOAD REGULATOR										
TWT Power Supply	5 2/136 5	440	5 2/136 5	440	5 2/136 5	440	5 2/136 5	440	5 2/136 5	440
Command Converter	0 62/19 0	38	0 62/19 0	38	0 62/19 0	38	0 62/19 0	38	0 62/19 0	38
Telemetry Encoder Converter	0 48/16 2	36	0 48/16 2	36	0.48/16 2	36	0 48/16 2	36	0 48/16 2	36
Repeater Logic	0 1/10 0	38	0 1/10 0	38	0 1/10 0	38	0 1/10 0	38	0 1/10 0	38
PACE (or MACE) Converter	0 62/25 2	54			0 62/25 2	54				
Phase Shifter Driver	4 44/193 0	364								
Subtotal	11 46/399 90	970	6 4/181 7	552	7 02/206 9	606	6 4/181 7	552	6 4/181 7	552
TOTAL	25 99/787 6	1988	22 43/614 4	1666	22 32/617 6	1684	23 01/632 4	1695	24 26/670 4	1840

TABLE 5-2 EFFICIENCY CHART FOR DC CENTRALIZED REGULATION DESIGN

	Synchronous' Altitude Spin-Stabilized ATS-B (F-1)				Medium Altitude Gravity Gradient ATS-A (F-2)			
	Beginning of Life		End of Life		Beginning of Life		End of Life	
	Output Power	Power Dissipation	Output Power	Power Dissipation	Output Power	Power Dissipation	Output Power	Power Dissipation
POWER SUBSYSTEM								
Switching Regulators'	Designed (150w)				Designed (100w)			
Driver Circuit Trans Switch		3.94		3 94		2 16		2 16
Series' Loss (Harness, Relay, etc)		7 34		6 35		4.00		3 73
Shunt Loss (Current Sensor, etc)		6.60		6 20		6.60		6 20
TYPICAL SPACECRAFT LOAD								
TWT Power Supply (1)	14 0	4 30	14 0	4.30	14.0	4 30	14 0	4.30
Command Regulator (Converter Loss) (2)	8 3	1 42	8 3	1 42	8 3	1 42	8 3	1 42
Telemetry Encorder Regulator (Converter Loss) (2)	6.8	0 80	6 8	0 80	3 4	0 0	3 4	0 0
Telemetry Transmitter (2)	8 6		8.6		4.3		4 3	
Repeater (Logic Loss) (2)	9.3	0.40	4.65	0 20	4 65	0 20	4 65	0 20
PACE Regulator Converter Loss)	12 4	1 76	6 2	0 88				
Phase Shifter Driver	5.0	1 2	5 0	1.20				
Payload 1	30.0		30.0		14 4		14 4	
Payload 2	35.0		35.0		22.0		14 4	
Total Typical Load	129 4	27.76	118.55	25.29	71.05	19 08	63 45	18 41
Total Typical Load Current, amp	5.4				2 96		2.64	
Total Power Required	129 4 + 27 76 = 157 2		118 55 + 25.29 = 143 84		71.05 + 19.08 = 90 13		63 45 + 18 41 = 81 86	
Power Available	167.5w at 33V				101 w at 33V 9.2w			
Efficiency	82 3%		82 4%		78 8 %		77 5 %	

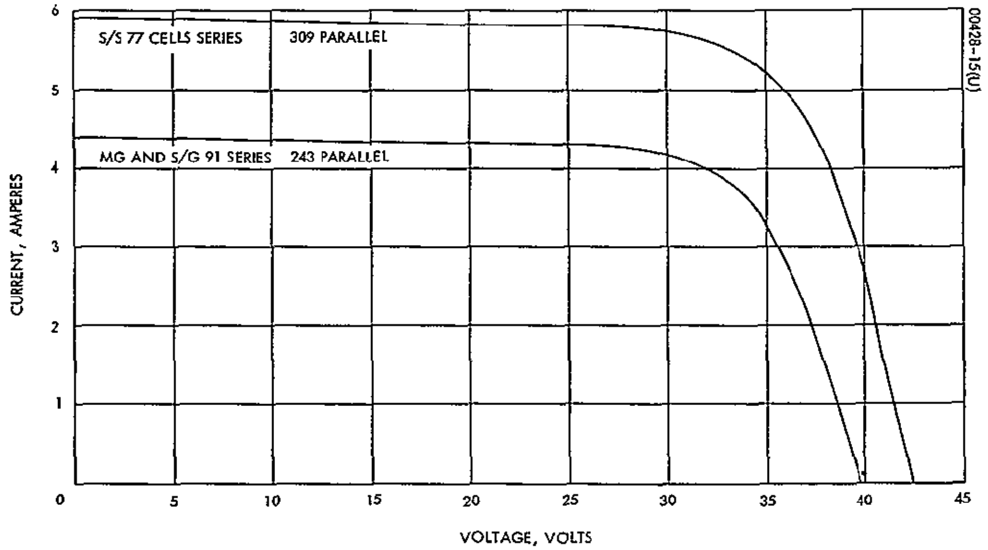


Figure 5-3. Reconfigured ATS Solar Panel EI Curves — Air Mass Zero, Beginning of Life

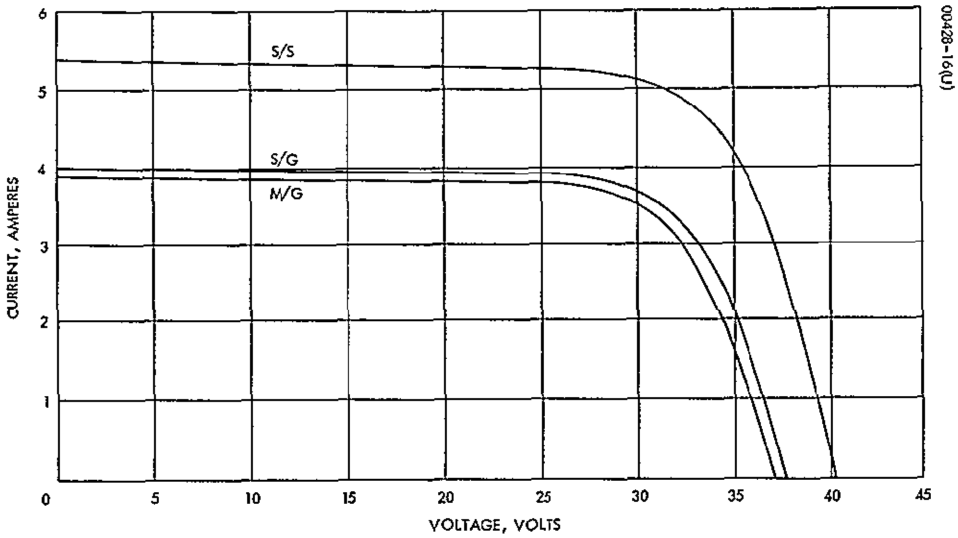


Figure 5-4. Reconfigured ATS Solar Panel EI Curves — End of Life

91 cells in series and S/S has 77 cells in series. The total of 243 strings are in parallel for S/G and M/G and 309 strings are in parallel for S/S. For the battery charge array, 12 strings are tapped from the main array for S/S and 13 and 25 strings are tapped from its main array for S/G and M/G, respectively. The solar array compositions for the system are shown in Table 5-3.

Battery

Battery design for the system is identical to the existing ATS design.

Current Sensors and Battery Discharge Unit

The total of eight current sensors are used to provide general housekeeping data for performance of the centralized system. Four sensors are used per bus and their functions are as follows.

One sensor is used for battery charging information and one for battery discharging information. Two sensors are used to input and output of the switching regulator to determine the total solar array current and the regulated bus current, respectively. The current sensors are of dc transformer type.

The battery discharge and charge unit has the diodes for battery charging and discharging. Additional diodes are used to interconnect between the regulated bus and a lower battery voltage. The purpose of this function is to furnish additional current required to blow the output fuses when the switching regulator is current limiting.

Switching Regulator and Input/Output Relays

Figures A-16, A-17, and A-18 (Appendix A) is a detailed shematic of the switching regulator for the centralized regulation system. It consists of input and output filters, a bias power supply, control circuitry, clocks, voltage and current amplifiers, driver and power transistors, and a commutating diode.

TABLE 5-3. MAIN SOLAR ARRAY

	S/S	S/G	M/G
Maximum power, watts	≈180	≈129	≈127
Rated voltage, volts	-33	-31	-31
Number of cells in series/strings	77	91	91
Number of strings	309	243	253
Number of strings used for battery charge	12	13	25

The switching regulator uses a duty cycle control system consisting of a bucking type switching regulator in which the unregulated bus voltage is chopped by a pass transistor switch and fed into an averaging filter. Regulated output voltage which is distributed to the spacecraft load and payload is maintained by controlling the switching duty cycle.

A dc error signal, filtered to obtain stability and remove transients, is added to an ac triangular waveform. This composite signal is compared to a reference with a comparator amplifier. The output of the comparator is a duty cycle modulated waveform.

During normal operations the current amplifier is saturated to its plus output. As the current feedback signal approaches the reference voltage, this amplifier becomes active. The output of this amplifier provides a signal to the voltage comparator which appears as an increase in output voltage. This, in turn, causes the voltage comparator to decrease on time and control the output current.

Overload and overvoltage protection are provided by means of fuses and a crowbar circuit respectively. In the event of a short at the input or output of the switching regulator, the fuses will blow to isolate a short. In case of a shorted pass transistor, the crowbar circuit activates blowing the input fuses. In addition, the regulated bus is connected to a lower battery voltage through diodes to provide the necessary current to blow the output fuses if a short occurs at the regulator output terminals.

The operation of input and output relay is similar to the ATS bus relay. A command to one of relay driver will connect the buses together and another command to another relay driver will disconnect the buses.

A separate current sensor is used in the regulator to provide telemetry data for each regulated bus and also to provide a signal to the current amplifier.

The bias power supply is of the series dissipative type. It provides auxiliary power for the regulator.

Table 5-4 lists the main performance characteristics of the centralized regulation system.

Reliability

Reliability analysis was performed on the dc centralized regulation system. The purpose of this analysis is to compare the reliability of two systems, the dc centralized regulation system and the existing ATS decentralized system. To maintain consistency, reliability criteria used for this analysis are the same as those used in previous modified ATS power subsystem. The reliability logic diagram for the centralized

TABLE 5-4. DC CENTRALIZED REGULATION DESIGN PERFORMANCE CHARACTERISTICS

Solar array	
Forward Aft	Basically identical to existing ATS design except for nominal voltage is approximately 33 volts.
Battery charge arrays	12 strings of main array-diode isolated.
Batteries	Same as existing ATS design; two-six amp-hr 22 cell batteries
Current sensors	8 required
Input and output relay	Similar to the existing ATS design.
Switching regulator	2 required
Input voltage from panel	-25 to -56 volts
Output voltage to bus	-24 volts
Regulation	±1 percent
AC ripple	50 mv p-p for spacecraft load 10 mv p-p for payload
Power capacity	150 watts maximum per regulator
Overload protection	
Input and output fuses	3 times total load per bus
Load and filter fuses	2 times load
Overvoltage protection	-30 volts
Efficiency	Approximately 88 percent
Battery discharge current Equalization (between 2 batteries)	40/60 percent unbalance maximum.

Table 5-4 (continued)

On/off control for loads	
Load relay	Average 11 per bus
Load isolation filters	Average 11 per bus
Weight and size	
Battery discharge units	2.08 lb/49 in^3
Input/output relays	0.55 lb/16.7 in^3
Switching regulators	6.4 lb/155 in^3
Load relay and filter	5.5 lb/167 in^3
Reliability	0.9752

power subsystem is shown in Figure 5-5 and a summary of centralized power subsystem reliability prediction and unit reliability predictions is shown in Appendix B.

The conclusion of this analysis is that the dc centralized regulation system has a reliability of 0.97520. This reliability assessment has no reliability "experience" factor included in part failure rates. If recommended experience factor of 0.512 is applied, the system reliability changes to approximately 0.98730.

CENTRALIZED CONVERSION DESIGN

The centralized conversion system is similar to the centralized regulator described in the first part of this section and is also designed to meet all the ATS requirements listed in Section 2. The centralized conversion design is implemented by addition of redundant inverters distributing a 50 volt square wave bus to the using systems. Voltage, waveform, and harness layout for EMI and efficiency optimization were not considered. The potential changes caused by additional optimization have a minimum influence on the comparative data.

The redundant inverters operate directly from the regulated bus. A block diagram of the system is shown in Figure 5-6. The system is identical to the centralized regulation system with the addition of the inverters, automatic transfer device, and transformer rectifiers at the load. These transformer rectifiers replace the dc to dc converters located internal to the load in the present system.

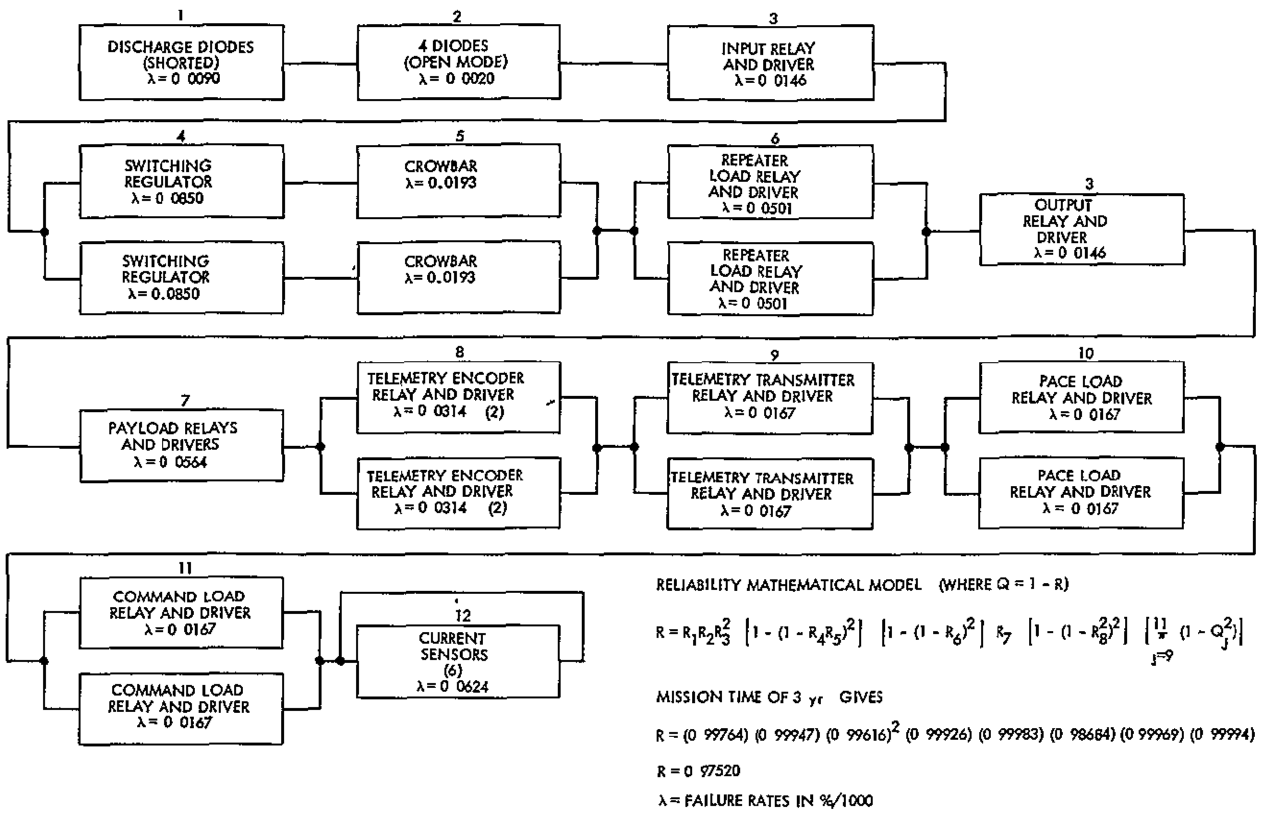


Figure 5-5. Centralized Power Subsystem Reliability Logic Diagram

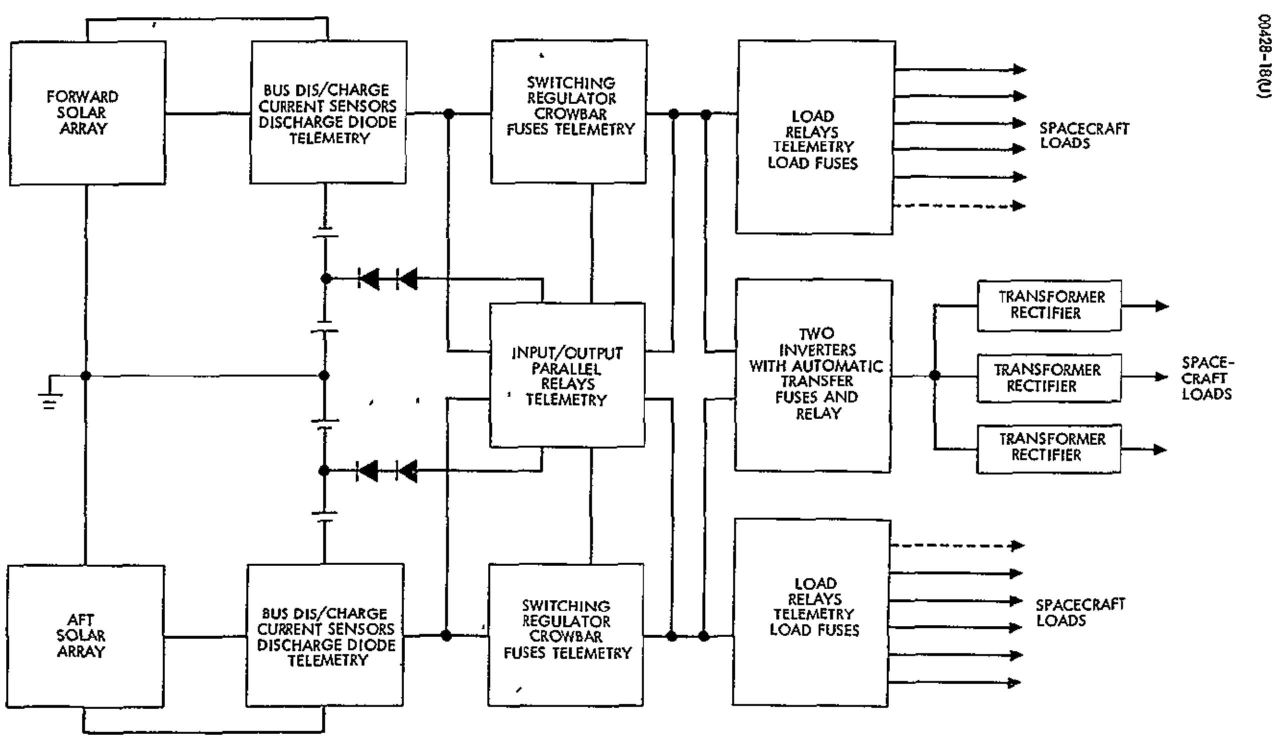


Figure 5-6. ATS Centralized Conversion Functional Block Diagram

Design data for the inverters and transfer relay have been added to the tabulations previously shown for the centralized system. Weight, volume, and parts count for the five ATS spacecraft are presented in Table 5-5 and output power, power dissipation, and efficiency for ATS-A and -B for both beginning and end of life are presented in Table 5-6. The estimate efficiency of the inverters is 91 percent. Other data is as shown. A comparison of this data with that of the other systems is given in Section 6.

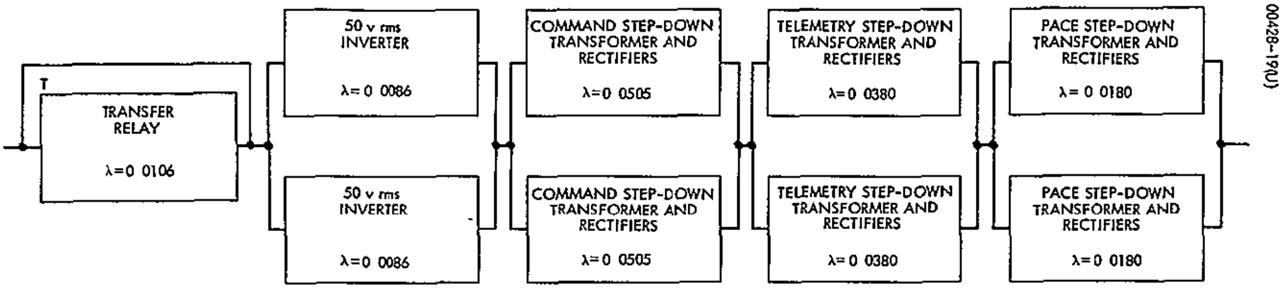
A schematic of the inverter circuit is provided in Figure A-21. The inverter is a conventional saturating transformer square wave oscillator with a self-starting driver circuit. The input to the inverter is coupled to both buses through fuses and relay contacts. The second inverter is automatically connected if the first inverter fails to supply power.

A reliability comparison of the redundant inverters utilizing remote transformer rectifiers and the present remote dc to dc converters can be seen by referring to Figures 5-7 and 5-8. The data shows values of 0.99969 and 0.99982, respectively indicating negligible difference between systems. Thus, reliability is not an important consideration in the comparison of centralized versus decentralized conversion for the ATS program. Data supporting these predictions is given in Appendix B.

CENTRALIZED AC DISTRIBUTION SYSTEM

This system is designed from building blocks used in the centralized systems previously described and is shown in Figure 5-9. Regulation of the ac voltage is provided by sensing the inverter output voltage and controlling the dc output of the switching regulator accordingly. In this system the spacecraft loads on both buses are always powered from one inverter and switching regulator. The panels are paralleled. If a failure of an inverter or a switching regulator occurs, power is automatically switched to the other inverter circuit. Detailed designs for all circuits associated with this system were not completed in this study. However, the data provided is suitable for comparative purposes. For this system, the weight, volume, parts count, output power, dissipation, and efficiency were tabulated for ATS-B only. This data is presented in Tables 5-7 and 5-8. A reliability assessment for this sytem was not made.

Schematics of the key power and control circuits are given in Appendix A.



RELIABILITY MATHEMATICAL MODEL (WHERE Q = 1 - R)

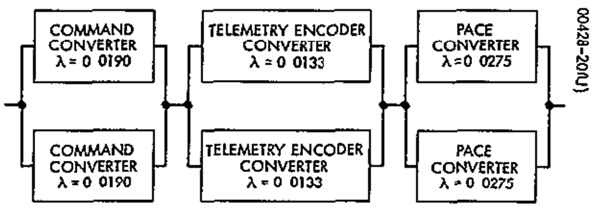
$R = (1 - Q_1^2)(1 - Q_2^2)(1 - Q_3^2)(1 - Q_4^2)$

MISSION TIME OF 3 yr GIVES RELIABILITY OF

R = (0 99999) (0 99982) (0 99990) (0 99998)

= 0 99969

Figure 5-7. Centralized Conversion Reliability Logic Diagram



RELIABILITY MATHEMATICAL MODEL (WHERE Q = 1 - R)

$R = (1 - Q_1^2)(1 - Q_2^2)(1 - Q_3^2)$

MISSION TIME OF 3 yr GIVES RELIABILITY OF

R = (0 99988) (0 99998) (0 99995)

= 0 99982

Figure 5-8 Decentralized Conversion Reliability Logic Diagram

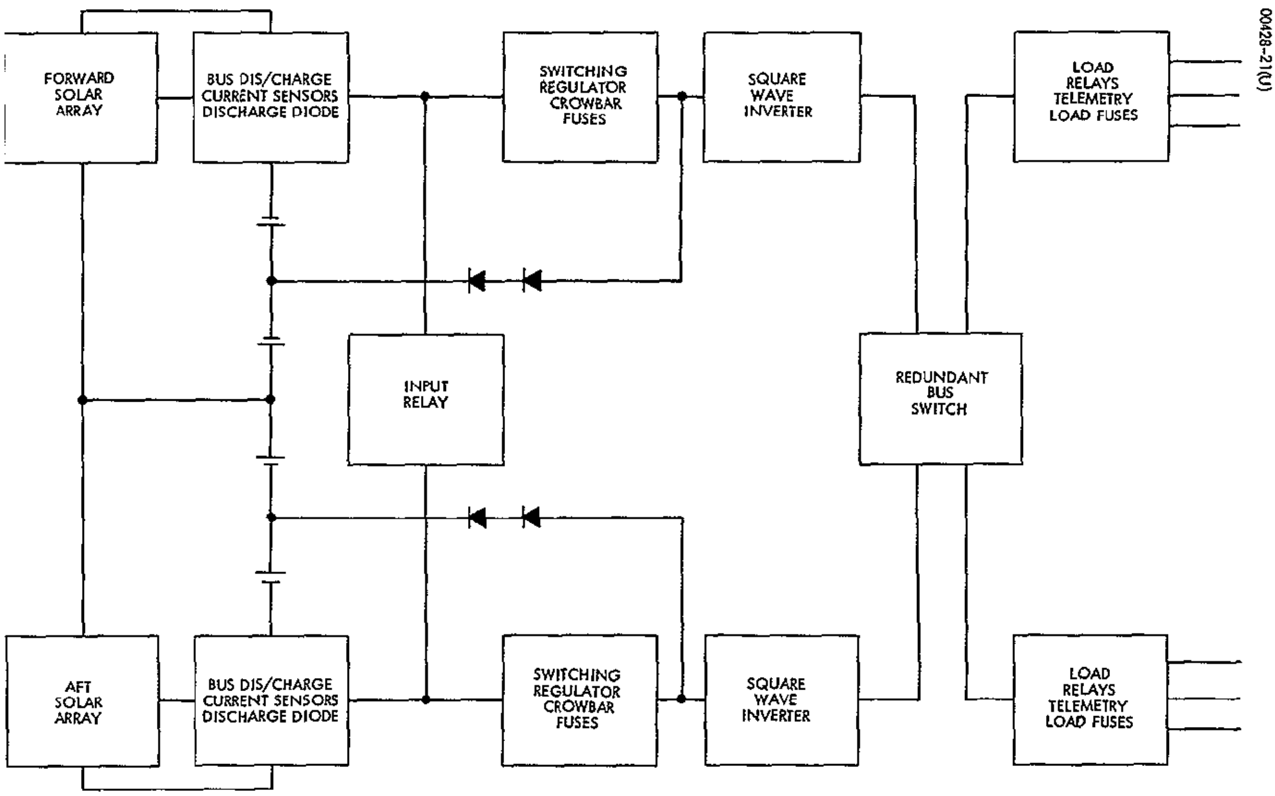


Figure 5-9. ATS Centralized AC Distribution System

TABLE 5-5 WEIGHT/VOLUME/PARTS COUNT CHART FOR CENTRALIZED CONVERSION DESIGN

	Synchronous Altitude Spin-Stabilized ATS-B (F-1)		Medium Altitude Gravity Gradient ATS-A (F-2)		Synchronous Altitude Spin-Stabilized ATS-C (F-3)		Medium Altitude Gravity Gradient ATS-D (F-4)		Synchronous Altitude Gravity Gradient ATS-E (F-5)	
	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count	Weight/ Volume, lb/in^3	Parts Count
POWER SUBSYSTEM										
Battery Discharge Unit with current sensors and Telemetry	2 08/49	82	2 08/49	82	2 08/49	82	2 08/49	82	2 08/49	82
Input/Output Relay	0 55/16 7	40	0 55/16 7	40	0.55/16 7	40	0.55/16 7	40	0 55/16 7	40
Switching Regulators	6 4/155	230	6 4/155	230	6 4/155	230	6 4/155	230	6.4/155	230
Load Relays and Filters	5 5/167	666	7.0/212	760	6 27/190	726	7 58/230	791	8 83/268	936
Inverters and Relay	1 0/30 4	46	1 0/30 4	46	1 0/30 4	46	1 0/30 4	46	1.0/30 4	46
SPACECRAFT LOAD REGULATOR										
TWT Power Supply	5 2/136 5	440	5 2/136 5	440	5 2/136 5	440	5 2/136 5	440	5 2/136 5	440
Command Transformer Rectifier	0 30/9 2	18	0.30/9 2	18	0 30/9 2	18	0.30/9 2	18	0 30/9 2	18
Telemetry Encorder Transformer-Rectifier	0 38/12 8	20	0 38/12 8	20	0 38/12 8	20	0 38/12 8	20	0 38/12 8	20
Repeater Logic	0 1/10 0	38	0 1/10 0	38	0 1/10 0	38	0 1/10 0	38	0.1/10 0	38
PACE (or MACE) Transformer-Rectifier	0.32/13.0	46			0.32/13.0	46				
Phase Shifter Driver	4 44/193 0	364								
TOTAL	26 27/792 6	1990	23 01/631 6	1674	22.6/622 6	1686	23 59/649 6	1705	24 84/687 6	1850

TABLE 5-6 EFFICIENCY CHART FOR CENTRALIZED CONVERSION DESIGN

	Synchronous' Altitude Spin-Stabilized ATS-B (F-1)				Medium Altitude Gravity Gradient ATS-A (F-2)			
	Beginning of Life		End of Life		Beginning of Life		End of Life	
	Output Power	Power Dissipation	Output Power	Power Dissipation	Output Power	Power Dissipation	Output Power	Power Dissipation
POWER SUBSYSTEM								
Switching Regulators'	Designed 150w				Designed 100w			
Driver Circuit		3 94		3 94		2 16		2 16
Series loss (transformer switch, harness, relay)		7 34		6 35		4 00		3 73
Shunt loss (current sensor)		6 60		6.20		6 60		6 20
Inverter loss		0 70		0.70		0 70		0 70
TYPICAL SPACECRAFT LOAD								
TWT Power Supply (1)	14 0	4 30	14.0	4 30	14 0	4 30	14 0	4 30
Command Regulator Transformer Regulator Loss (2)	8 3	0 69	8 3	0 69	8 3	0 69	8 3	0 69
Telemetry Encorder Regulator (Regulator Loss)(2)	6 8	0 43	6 8	0.43	3.4	0.23	3 4	0 23
Telemetry Transmitter(2)	8 6		8.6		4 3		4 3	
Repeater (Logic Loss) (2)	9 3	0 40	4 65	0 20	4 65	0 20	4 65	0 20
PACE Regulator (Transformer Regulator Loss)	12 4	0 89	6 2	0.44				
Phase Shifter Driver	5.0	1 2	5.0	1 20				
Payload 1	30 0		30.0		14 4		14 4	
Payload 2	35 0		35 0		22.0		14 4	
Total Typical Load	129 4	26 49	118 55	24.45	71 05	18 88	63 45	18.21
Total Typical Load Current, amp	5 4				2.96		2 64	
Total Power Required	129 4 + 26.29 = 155 89		118 55 + 24 25 = 143.0		71.05 + 18 88 = 89 93		63.45 + 18 21 = 81.86	
Power Available	*167 5w at 33V*				*101w at 33V*			
Efficiency	83 0%		82 9%		79 0%		77 5%	

TABLE 5-7. WEIGHT/VOLUME/PARTS COUNT CHART FOR AC CENTRALIZED DISTRIBUTION DESIGN

	Synchronous Altitude Spin Stabilized ATS-B (F-1)	
	Weight, Volume, lb/in^3	Parts Count
POWER SUBSYSTEM		
Battery Discharge Unit with Current Sensors and Telemetry	2.08/49	94
Redundant Bus Relay	0.3/10.0	36
Input Relay	0.25/8.3	18
Switching Regulators	6.4/155	238
Load Relays and Filters	5.5/167	666
Inverters and Transformers	2.46/75.0	132
SPACECRAFT LOAD REGULATOR		
TWT Power Supply	5.2/136.5	440
Command Transformers - Rectifiers	0.40/12.2	52
Telemetry Encoder Transformer Rectifier	0.58/19.5	30
Repeater Transformer	0.4/40.0	48
PACE (or MACE) Transformer Rectifier	0.33/14.0	58
Phase Shifter Driver	4.44/193.0	364
Telemetry Transmitter Sublaunching Solid	0.76/25.6	12
TOTAL	29.09/905.1	2185

TABLE 5-8. EFFICIENCY CHART FOR AC CENTRALIZED DISTRIBUTION DESIGN

	Synchronous Altitude Spin-Stabilizer ATS-B (F-1) at BOL	
	Output Power	Power Dissipation
POWER SUBSYSTEM		
Switching Regulators	Designed 150w	
Driver Circuit		3.94
Series loss (harness, transmitter, switch, relay prop, etc.)		7.34
Shunt loss (current sensors, etc.,)		6.60
Inverter loss		13.00
TYPICAL SPACECRAFT LOAD		
TWT Power Supply	14.0	4.30
Command Regulator (Transformer-Rectifier Loss)	8.3	0.69
Telemetry Encoder Regulator (Transformer-Rectifier Loss)	6.8	0.43
Telemetry Transmitter Regulator (Transformer-Rectifier Loss)	8.6	
Repeater (Logic Loss)	9.3	0.40
PACE Regulator (Transformer-Rectifier Loss)	12.4	0.89
Phase Shifter Driver	5.0	1.20
Payload 1	30.0	
Payload 2	35.0	
Total Typical Spacecraft Load	129.4	38.79
Efficiency	77%	

6. DECENTRALIZED VERSUS CENTRALIZED SYSTEM COMPARISONS

This study provided the opportunity of taking a second look at the power systems used on five different ATS spacecraft to determine the trade-offs associated with other configurations. Comparison of data of the selected configurations is presented in this section. In presenting this comparative data, several limitations of the study should be pointed out.

1) The ATS power system interfaced with GFE experiments through the payload regulators which are part of the ATS power system. The additional regulation and conversion required internal to the GFE experiments were not considered in this study. While it was recognized that this data was of interest, sufficiently detailed data was not available.

2) Comparisons are made between a fully designed system and a conceptual design. It is expected that some of the assumptions made for the conceptual system may not be completely valid. Also the circuit designs are expected to require modification and incorporation of additional circuitry during the development program.

In spite of these limitations, the data comparison is of great interest and provides answers to questions of vital concern to the power system designer. This comparative data is presented in the following summary tables:

1) Table 6-1 - Solar Array Configuration Comparison

2) Table 6-2 - Parts Count Summary for Existing ATS System and DC Centralized Regulation System

3) Table 6-3 - Weight/Volume Comparison Chart for Existing ATS System and DC Centralized Regulation System

4) Table 6-4 - Parts Count Summary for Existing ATS System and Centralized Conversion System

5) Table 6-5 - Weight/Volume Comparison Chart for Existing ATS System and Centralized Conversion System

6) Table 6-6 - Cost Comparison -- Present ATS System Versus Centralized Regulation System

Of this data, Tables 6-1, 6-2, and 6-3 are the most significant. Tables 6-4 and 6-5 show very little change from Tables 6-2 and 6-3, demonstrating that for the ATS system the maximum change occurs when going to a centralized regulation system. Use of centralized conversion has negligible additional effect on parts count, size, and weight. Table 6-6 discusses cost qualitatively which is a difficult comparison to present accurately.

Table 6-1 compares solar panel designs which were based on using the identical panel area. The present ATS design uses charge arrays in series with the main array for battery charging. The new design uses 12, 13, or 25 (depending on the spacecraft) full length strings (77 cells in ATS-B, for example) paralleled through isolation diodes for battery charging. The interesting comparison between the two systems is that during battery charging, the main panel power available is identical for both systems. There are two major differences in panel utilization, however. One is due to the different layout described above. In the present ATS system if a switch were added to turn off battery power, the charge array power would not be available to the load. In the new design switching off battery charge would permit the charge power to be automatically available to the load.

A second major difference in the new panel design and centralized regulator is the power available to the load at the beginning of life. This can be seen from the table to be 9 watts for the synchronous satellites and 6.0 watts for the other spacecraft. No power difference exists at end of life.

Table 6-2 shows a substantial part reduction for the centralized regulation system. Allowing for some parts growth during development, a fully implemented centralized system would still be expected to use substantially fewer parts than the present ATS parts complement. The parts delta between designs is approximately the same for all spacecraft except the ATS-E which uses 950 parts less than the present design. This is due mainly to the need for a switching converter, the use of six limiters, and the high number of payload regulators to power the experiments.

The weight comparison shown in Table 6-3 is also of considerable interest. It shows that, for ATS, the present design is weight effective. This is somewhat surprising because of the many fairly complex circuits eliminated by the change to the centralized system. This includes the limiters, battery controllers, and many series regulators. Most of these units, however, are light and the weight of the redundant switching regulators and load relays and filters is greater than that of the deleted units. This weight trend holds for all spacecraft except ATS-E. This again is due to the switching converters

TABLE 6-1 SOLAR ARRAY CONFIGURATION COMPARISON

Season - Equinox
Summer Solstice Solar Panel Current Reduced by 11.2%

Characteristic	Synchronous Attitude Spin Stabilized ATS-B & -C (F-1 & -3)				Synchronous Attitude Gravity Gradient ATS-D & -E (F-4 & F-5)				Medium Attitude Gravity Gradient ATS-A (F-2)			
	Present Decentralized System		DC Centralized System		Present Decentralized System		DC Centralized System		Present Decentralized System		DC Centralized System	
	BOL	EOL	BOL	EOL	BOL	EOL	BOL	EOL	BOL	EOL	BOL	EOL
Maximum Main Solar Panel Current, amps	6.69 @ 26.9 V	6.15 @ 25.5 V	5.5 @ 33 V	5.05 @ 31 V	4.53 @ 27.5 V	4.28 @ 25.5 V	3.9 @ 33 V	3.55 @ 31 V	4.53 @ 27.5 V	4.07 @ 25.5 V	3.9 @ 33 V	3.35 @ 31 V
Number of Cells in Series String (Main Solar Panel)	62	62	77	77	80	80	91	91	80	80	91	91
Number of Strings in Parallel	378	378	309	309	270	270	243	243	270	270	243	243
Total Number of Cells	23,796	23,796	23,793	23,793	21,864	21,864	22,113	22,113	22,128	22,128	22,113	22,113
Battery Charge Current, amp	0.212	0.195	0.213	0.196	0.201	0.190	0.208	0.19	0.402	0.362	0.402	0.345
Number of Cells in Series String (Charge Array)	15	15	77	77	11	11	91	91	11	11	91	91
Number of Strings in Parallel	12	12	12	12	12	12	13	13	24	24	25	25
Current Available in Main (Solar Panel for Spacecraft Load), amp	6.256	5.75	5.08	4.66	4.13	3.90	3.38	3.17	3.63	3.34	3.06	2.66
Main Solar Panel Power, w	168	146	167.5	145	113.5	99.4	114.8	98.2	100	85.0	101.0	82.5
Battery Charge Solar Panel Power, w	12	11	14.0	12.1	12	9.6	14	11.8	24.8	18.8	26.5	21.4
Efficiency of System, %	77.2	82.2	82.3	82.4		Similar to M/G ATS-A (F-1)			73.45	78.1	78.8	77.5
Power Available for Spacecraft, w	129	120	138	≈120					73.5	66.3	79.5	63.9
Δ Power Available, w			9								6.0	

TABLE 6-2 PARTS COUNT SUMMARY FOR EXISTING ATS SYSTEM AND DC CENTRALIZED REGULATION SYSTEM

Control Item	Synchronous Attitude Spin Stabilized ATS-B (F-1)		Medium Attitude Gravity Gradient ATS-A (F-2)		Synchronous Attitude Spin Stabilized ATS-C (F-3)		Synchronous Attitude Gravity Gradient ATS-D (F-4)		Synchronous Attitude Gravity Gradient ATS-E (F-5)	
	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System
POWER SUBSYSTEM (by Unit)										
Battery Discharge Control	152	82	152	82	152	82	152	82	152	82
Bus Relay	45	40	45	40	45	40	45	40	45	40
Current Sensor	108	in battery	108	in battery	108	in battery	108	in battery	108	in battery
Voltage Limiter	100	-	150	-	100	-	150	-	150	-
Current Control Unit	-	-	34	-	-	-	34	-	34	-
Switching Regulator (or Converter)	-	230	-	230	-	230	-	230	205	230
Load Relay and Filter	-	666	-	762	-	726	-	791	-	936
SPACECRAFT LOAD REGULATOR										
TWT Power Supply	440	440	440	440	440	440	440	440	440	440
Command (dc-dc Converter)	112	38	112	38	112	38	112	38	112	38
Telemetry Encoder (dc-dc Converter)	196	36	196	36	196	36	196	36	196	36
Telemetry Transmitter	176	-	176	-	88	-	88	-	176	-
Repeater (Logic)	244	38	244	38	244	38	244	38	244	38
PACE (MACE) (dc-dc Converter)	180	54	-	-	180	54	-	-	-	-
Subliming Solid Driver	-	-	246	-	-	-	246	-	246	-
Phase Shifter Driver	364	364	-	-	-	-	-	-	-	-
Subtotal										
EXPERIMENT PAYLOAD REGULATOR	344	-	275	-	620	-	496	-	682	-
Total	2461	1988	2178	1666	2285	1684	2311	1695	2790	1840
Δ Parts Count	473		512		601		616		950	

TABLE 6-3 WEIGHT/VOLUME COMPARISON CHART FOR EXISTING ATS SYSTEM AND DC CENTRALIZED REGULATION SYSTEM

Weight/Volume (lb/in) of Existing ATS Decentralized System Actual.
Weight/Volume of Designed DC Centralized System Estimated.

Control Item	Synchronous Attitude Spin Stabilized ATS-B (F-1)		Medium Attitude Gravity Gradient ATS-A (F-2)		Synchronous Attitude Spin Stabilized ATS-C (F-3)		Synchronous Attitude Gravity Gradient ATS-D (F-4)		Synchronous Attitude Gravity Gradient ATS-E (F-5)	
	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System
POWER SUBSYSTEM										
Battery Discharge Unit	2 2/51 8	2 08/49	2 2/51 8	2 08/49	2 2/51 8	2 08/49	2 2/51 8	2 08/49	2 2/51 8	2 08/49
Current Sensor	1 20/40		1 2/40		1 2/40		1 2/40		1 2/40	
Bus Relay	0 34/10 3	0 55/16 7	0 34/10 3	0 55/16 7	0 34/10 3	0.55/16 7	0 34/10 3	0 55/16 7	0 34/10 3	0 55/16 7
Voltage Limiter	2 48/65 5	-	3 72/98 5	-	2 48/65 5	-	3 72/98 5	-	3 72/98 5	-
Current Control Unit	-	-	0 13/5 08	-	-	-	0 13/5 08	-	0 13/5 08	-
Switching Regulator	-	6 4/155 0	-	6.4/155 0	-	6 4/155 0	-	6 4/155 0	5 56/123 5	6 4/155 0
Load Relays and Filter	-	5 5/387 7	-	7 0/212	-	6 27/190	-	7 58/230	-	8 83/268
SPACECRAFT LOAD REGULATOR										
TWT Power Supply	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5	5 2/136 5
Command	1 0/30 8	*0 62/19 0	1 0/30 8	*0 62/190	1 0/30 8	*0 62/19 0	1 0/30 8	*0 62/19 0	1 0/30 8	*0 62/19 0
Telemetry Encoder	1 0/33 8	*0 48/16 2	1 0/33 8	*0 48/16 2	1 0/33 8	*0 48/16 2	1 0/33 8	*0 48/16 2	1 0/33 8	*0 48/16 2
Telemetry Transmitter	1 2/107 2	-	1 2/107 2	-	0 6/53 6	-	0 6/53 6	-	1 2/107 2	-
Repeater (or Logic)	0 62/53 6	0 1/10 0	0 62/53 6	0 1/10 0	0 62/53 6	0 1/10 0	0 62/53 6	0 1/10 0	0 62/53 6	0 1/10 0
PACE or MACE	1 0/40 6	*0 62/25 2	-	-	*1 0/40 6	0 62/25 2	-	-	-	-
Subliming Solid Driver	-	-	0 912/37 2	-	-	-	0 912/37 2	-	0 912/37 2	-
Phase Shifter Driver	4 44/193 0	4 44/193 0	-	-	-	-	-	-	-	-
Experiment Payload Regulator	1 86/76 8	-	1 46/64	-	3 06/128	-	2 27/102	-	3 23/141	
Total	22 54/783 9	25 99/787 6	18 98/612 78	22 43/614 4	18 7/616 5	22 32/617 6	19 19/625 18	23 01/632 4	26 31/813 28	24 26/67 04
Δ	+ 3 45/3 70		+ 3 45/1 62		+ 3 62/1 10		+ 3 82/7 22		- 2 05/142 88	

*Weight of dc-dc converter

TABLE 6-4 PARTS COUNT SUMMARY FOR EXISTING ATS SYSTEM AND CENTRALIZED CONVERSION SYSTEM

Control Item	Synchronous Attitude Spin Stabilized ATS-B (F-1)		Medium Attitude Gravity Gradient ATS-A (F-2)		Synchronous Attitude Spin Stabilized ATS-C (F-3)		Synchronous Attitude Gravity Gradient ATS-D (F-4)		Synchronous Attitude Gravity Gradient ATS-E (F-5)	
	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System
POWER SUBSYSTEM (or Unit)										
Battery Discharge Control	152	82	152	82	152	82	152	82	152	82
Bus Relay	45	40	45	40	45	40	45	40	45	40
Current Sensor	108	in battery	108	in battery	108	in battery	108	in battery	108	in battery
Voltage Limiter	100	-	150	-	100	-	150	-	150	-
Current Control Unit	-	-	34	-	-	-	34	-	34	-
Switching Regulator (or Converter)	-	230	-	230	-	230	-	230	205	230
Load Relay and Filter	-	666	-	760	-	726	-	791	-	936
Inverters and Relay	-	46	-	46	-	46	-	46	-	46
SPACECRAFT LOAD REGULATOR										
TWT Power Supply	440	440	440	440	440	440	440	440	440	440
Command Transformer-Rectifier	112	18	112	18	112	18	112	18	112	18
Telemetry Encoder (Transformer-Rectifier)	196	20	196	20	196	20	196	20	196	20
Telemetry Transmitter	176	-	176	-	88	-	88	-	176	-
Repeater (Logic)	244	38	244	38	244	38	244	38	244	38
PACE (MACE) (Transformer-Rectifier)	180	46	-	-	180	46	-	-	-	-
Subliming Solid Driver	-	-	246	-	-	-	246	-	246	-
Phase Shifter Driver	364	364	-	-	-	-	-	-	-	-
EXPERIMENT PAYLOAD REGULATOR	344	-	275	-	620	-	496	-	682	-
Total	2461	1990	2178	1674	2285	1686	2311	1705	2790	1850
Δ Parts Count	471		504		599		606		940	

TABLE 6-5 WEIGHT/VOLUME COMPARISON CHART FOR EXISTING ATS SYSTEM AND CENTRALIZED CONVERSION SYSTEM

Weight/Volume (lb/in) of Existing ATS Decentralized System Actual.
Weight/Volume of Designed Centralized Conversion System Estimated.

Control Item	Synchronous Attitude Spin Stabilized ATS-B (F-1)		Medium Attitude Gravity Gradient ATS-A (F-2)		Synchronous Attitude Spin Stabilized ATS-C (F-3)		Synchronous Attitude Gravity Gradient ATS-D (F-4)		Synchronous Attitude Gravity Gradient ATS-E (F-5)	
	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System	Present System	Centralized System
POWER SUBSYSTEM										
Battery Discharge Unit	2.2/51.8	2.08/49	2 2/51 8	2.08/49	2.2/51 8	2.08/49	2 2/51 8	2 08/49	2 2/51.8	2.08/49
Current Sensor	1.20/40	-	1 2/40	-	1 2/40	-	1.2/40	-	1 2/40	-
Bus Relay	0.34/10.3	0.55/16.7	0 34/10.3	0 55/16.7	0 34/10.3	0 55/16.7	0.34/10 3	0 55/16 7	0.34/10 3	0 55/16 7
Voltage Limiter	2 48/65.5	-	3.72/98.5	-	2.48/65.5	-	3 72/98.5	-	3 .72/98 5	-
Current Control Unit	-	-	0.13/5 08	-	-	-	0.13/5.08	-	0.13/5 08	-
Switching Regulator	-	6.4/155.0	-	6 4/155.0	-	6 4/155.0	-	6 4/155.0	5 56/123 5	6 4/155.0
Load Relays and Filter	-	5.5/387 7	-	7.0/212	-	6.27/190	-	7.58/230	-	8.83/268
Inverter and Relay	-	1.0/30.4	-	1.0/30 4	-	1.0/30.4	-	1.0/30 4	-	1.0/30 4
SPACECRAFT LOAD REGULATOR										
TWT Power Supply	5.2/136.5	5.2/136.5	5 2/136.5	5.2/136 5	5.2/136.5	5.2/136.5	5 2/136 5	5.2/136.5	5.2/136 5	5.2/136.5
Command	1.0/30 8	*0.30/9.2	1.0/30.8	*0.30/9.2	1.0/30 8	*0.30/9 2	1 0/30.8	*0.30/9 2	1 0/30 8	*0.30/9.2
Telemetry Encoder	1.0/33.8	*0.38/12.8	1.0/33 8	*0 38/12 8	1.0/33.8	*0.38/12.8	1.0/33.8	*0.38/12 8	1.0/33.8	*0 38/12 8
Telemetry Transmitter	1 2/107.2	-	1.2/107.2	-	0 6/53.6	-	0 6/53 6	-	1.2/107.2	-
Repeater (or Logic)	0 62/53.6	0.1/10.0	0.62/53.6	0.1/10.0	0.62/53.6	0 1/10 0	0 62/53.6	0.1/10.0	0 62/53 6	0.1/10.0
PACE or MACE	1 0/40.6	*0.32/13.0	-	-	*1.0/40.6	*0.32/13.2	-	-	-	-
Subliming Solid Driver	-	-	0.912/37.2	-	-	-	0.912/37.2	-	0 912/37.2	-
Phase Shifter Driver	4.44/193.0	4 44/193 0	-	-	-	-	-	-	-	-
Experiment Payload Regulator	1.86/76.8	-	1.46/64	-	3.06/128	-	2.27/102	-	3 23/141	-
Total	22.54/783 9	26 27/792 6	18.98/612.78	23.01/631.6	18.7/616.5	22.6/622 6	19.19/675.18	23.59/649.6	26.31/813.28	24.84/687.6
△	+ 3 73/8.70		+ 4.03/18 82		+ 3.90/6.1		+ 4 40/74 42		- 1.47/125.68	

*Weight of transformer-recrifier

TABLE 6-6 COST COMPARISON – PRESENT ATS SYSTEM VERSUS CENTRALIZED REGULATION SYSTEM

Task Description	Decentralized System (Present ATS)	Proposed Centralized System
Non-Recurring		
Design and Development	1 0	1 15
Qualification Testing (TV, Vibration, Electrical Performance and EMI)	1 0	1 0*
Recurring		
Unit Manufacturing, including Engineering support	1 0	1 1
Acceptance Testing (TV, Vibration and Electrical Performance)	1 0	1 0
Battery Discharge Control	2 (series regulator)	2 (3 diodes)
Current Sensor	8 (required 6 oscillators)	8 (required 4 oscillators)
Bus Relay	1 (with undervoltage sensor and relay)	2 (relay)
Bus Voltage Limiter	4	0
Switching Regulator	0	2 (regulator with a crowbar circuit and fuses)
TWT Power Supply	4 (series regulator, dc-dc converter, on/off driver	4 (series regulator, dc-dc converter, on/off driver
Command Regulator	2 (series regulator, dc-dc converter)	2 (load filter, fuses, dc-dc converter)
Telemetry Encoder Regulator	2 (2 series regulators, dc-dc converter, on/off driver	2 (2 load relay and filter, fuses, dc-dc converter)
Telemetry Transmitter Regulator	4 (series regulator, on/off driver)	4 (load relay and filter, fuses)
Repeater Regulator	2 (3 series regulator, on/off logic)	2 (3 load relays, and logic)
PACE Regulator	2 (series regulator, dc-dc converter, on/off driver)	2 (load relay, dc-dc converter)
Phase Shifter Driver	2 (8 voltage limiters and 8 dc-dc converter on/off)	2 (8 voltage limiters and 8 dc-dc converter on/off)
Payload Regulator	6 (series regulator on/off driver)	6 (load relay, and filter, fuses)
Relative Component Cost	1 0	0 75

*This relative number would be 1 4 if EMI testing was not required for the nonswitching type (present ATS) system, but was required for the centralized system

Changes in volume from one system to another are negligible, again with the exception of ATS-E.

The data analysis of the dc centralized design versus the present design described above is also applicable to the centralized conversion summaries shown in Tables 6-4 and 6-5. This results from the number of dc to dc converters which are capable of being replaced by the centralized converters. Two centralized redundant converters with automatic transfer and remotely located transformer rectifier units is approximately equivalent to the six dc-dc converters for three separate redundant loads used on the present ATS design. Even if the number of converters needed for the ATS system doubled, the two systems would still be roughly equivalent.

A table showing reliability comparisons is provided and discussed in Section 1. It shows improved reliability for the centralized regulation system but negligible difference for the centralized conversion system.

The factors affecting the cost can be broken into two main categories: recurring and nonrecurring. The nonrecurring portion includes system analysis, detail circuit design and development, and hardware qualification testing. The recurring portion includes the components, fabrication and assembly cost of the flight hardware, manufacturing support, and unit acceptance test (normally comprised of thermal vacuum, electrical performance, and vibration tests).

Because of the varied requirement of space programs, the comparison presented here is a relative comparison only with emphasis given where difference exists. In all cases, the present ATS decentralized system is assumed to have a unity value with the centralized system assigned a relative number above or below unity as the case may be. This is shown in Table 6-6. which also shows the realtive cost of the procured components (i.e., transistors, diodes, transformers, capacitors, resistors, etc.) and the unit hardware complement.

In summary, the centralized system would cost more than the decentralized system in both the nonrecurring and recurring categories. The added expense resulting from the added complexity of the power conversion units is minimal, however, and would not have a significant influence on the system selection.

7. POWER SYSTEMS DESIGN FACTORS AND REFERENCE DATA FOR FUTURE DESIGNS

A requirement of this study is to determine the factors that have a bearing on the centralization decision and to determine the extent to which these factors affect the degree of centralization to be used. Another requirement is to present the data so that future designs can be refrenced to it. This section combines these two requirements.

The approach to these requirements is to review all factors that affect the power systems design and then determine which factors have a major influence on the centralization decision. A brief description of each factor is provided here. The discussion is intended to summarize the typical influence of the factor on the power systems design and on the centralization decision. Several specific illustrations are given when helpful for clarification. Each factor is summarized as to importance to the centralization decision and with respect to the ATS spacecraft.

Figure 7-1 is a block diagram of the design factors that influence the power system design and the centralization decision. The centrally located boxes show the main path which must be followed in selection of the power subsystem. The other contributing factors are shown in the peripheral boxes. The tabulated data shown in the remainder of this section describe each box.

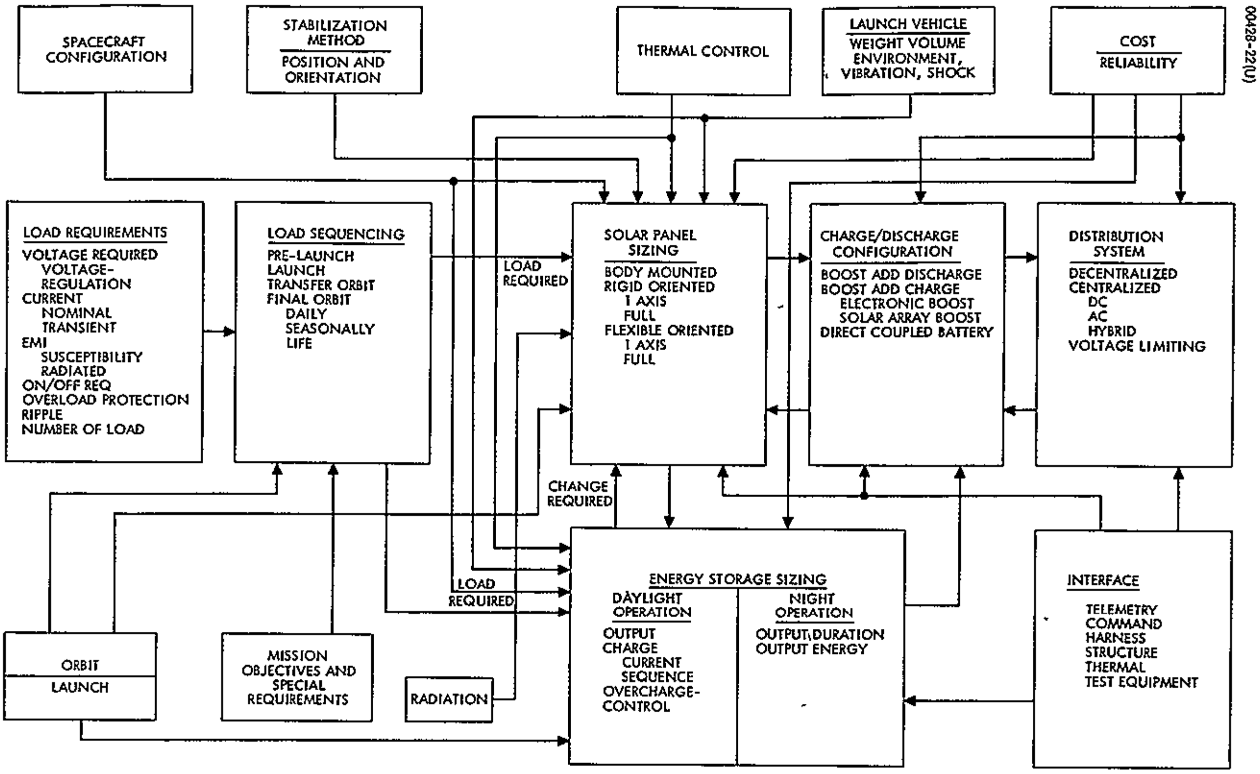


Figure 7-1 Factors That Affect Power Systems Design

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
1) Load Requirements	Load requirements are the starting point for the power systems design since they define the total power requirement of the system and the voltage levels required by the loads. The ATS load requirements are shown in Section 2. Load requirements have a major influence on the centralization decision with consideration given to the following categories.		
a) Voltage Regulation	Power system design can progress in several ways. The power system designer can impose either a regulated bus or an unregulated bus on the load user. In either case, absolute upper and lower voltage limits are provided. Figures 7-2 and 7-3 show EI curves for typical solar panel extremes and solar panel voltage versus time after emerging from eclipse in a synchronous orbit when no means of clamping the maximum panel voltage is provided. End of life bus voltage for this spacecraft is 24.0 volts. The power user must design his equipment to function reliably within whatever extremes the power system provides.	The need for precision regulation of the loads is a key factor in the centralized regulator decision.	With respect to the ATS spacecraft, precision regulation is required by only a small percentage of the loads, making centralized regulation a strong candidate system.

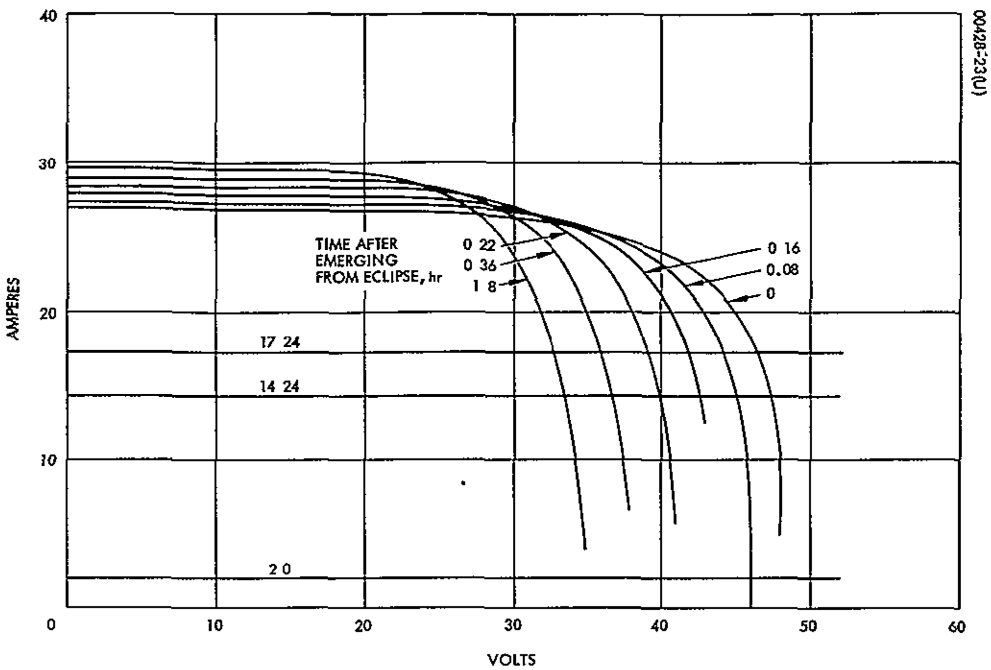


Figure 7-2. Panel EI Curves as Function of Time After Emerging from Eclipse

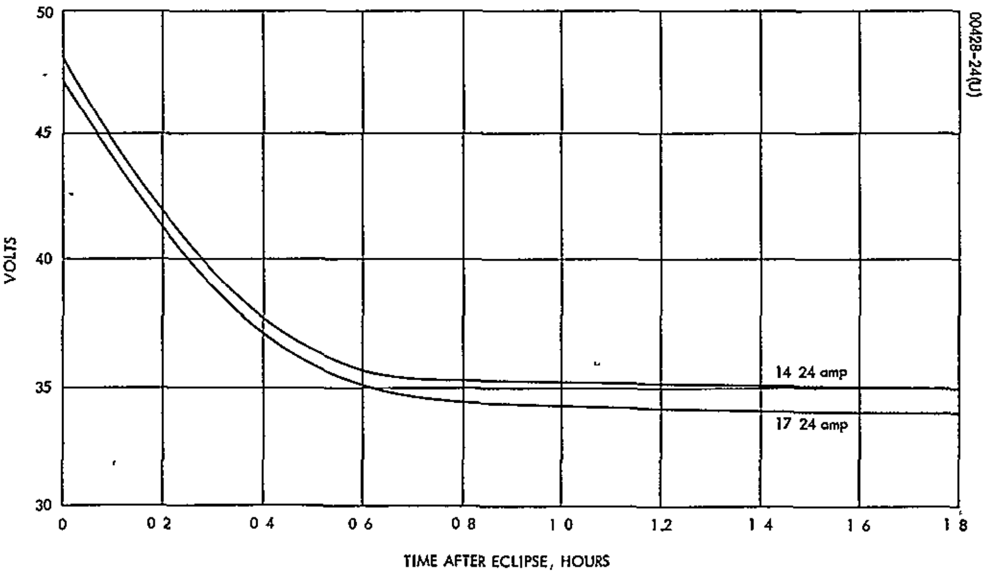


Figure 7-3. Solar Panel Voltage Versus Time After Eclipse for Two Current Levels

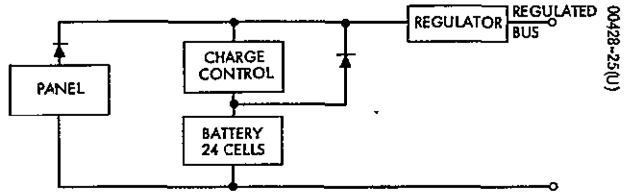


Figure 7-4 Regulated Bus System

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	It is sometimes more convenient to clamp the upper voltage limit with shunt regulators in order to reduce the stress on the load. The major tradeoff in this area is the simplification of the load design versus the unreliability of the limiters. A regulated bus can be provided by interposing a regulator between the panel and spacecraft bus, as shown in Figure 7-4. If this is done, it must be determined if the regulation is adequate for a high percentage of power usage. If a centralized regulation system is used for a spacecraft application, a regulator may still be needed remotely, internal to the load, if the load cannot operate satisfactorily with the accuracy provided by the centralized regulator. However, even if a remote regulator is needed, the centralized regulator may still provide other advantages (such as reduced dissipation, elimination of battery controller and limiters, improved power at beginning of life, etc.).		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	Voltage regulation is an important factor in the centralization decision. The need for precision voltage at the load negates the major advantage of a centralized system — namely, minimum parts count and maximum reliability. As was shown in this report, with respect to the ATS system, parts count can be reduced and reliability improved by going to a centralized system for a spacecraft with load requirements like that of the ATS. However, if most of the load requires its own regulators in addition in order to provide more precise regulation these advantages would vanish.		
b) Voltages Required	Much of the power provided by the spacecraft bus is used at a voltage level different from that of the bus. Positive and negative inputs (reversed polarity for one input) are needed for operational amplifiers. Low voltage is needed for the integrated circuits used in digital applications. TWT tubes, if used for communications, require both filament and high voltage supplies.	Centralized conversion has no advantages.	In the ATS design, only six dc to dc converters can be replaced by a centralized system. There is negligible difference in parts count and reliability in making this change.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	AC (inverters) are sometimes required for motor drives. Precision dc is required for digital to analog conversion, thermistor and pressure transducer excitation and calibration supplies. Requirements for experiments depend on the experiments but are probably typical of the above requiring positive and negative supplies, precision supplies, and high voltage supplies. The ATS spacecraft uses a number of dc to dc converters remotely located at the various loads. The regulator converter used for the TWT supplies cannot easily be centralized because each TWT regulator converter is matched to its own TWT tube. The remainder of the converters can be centralized. The comparative data presented in Section 6 shows no advantages for a centralized design.		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
c) Current Levels and Transient Load Requirements	Current levels determine the power each load requires. In a decentralized design, the bus voltage available is a variable. Either series dissipative regulators or switching regulators may be used to connect to the load. If series dissipative regulators are used, the load becomes a constant current load. If switching regulators are used, the load becomes a constant power load. One advantage of constant power loads is the ability to use the additional power available at the beginning of life before the solar panel has degraded. This is also true of constant current loads, but to a lesser degree. In the case of the ATS spacecraft, the decentralized regulators are dissipative. Use of a centralized switching regulator as concepted and described in this report would have permitted use of the experiments to a greater degree initially since the panel could have operated nearer its maximum power point. High load currents (or transients) during equipment turn-on, squib firing, or for other reasons, is another load requirement that must	Current levels and transient current requirements determine the power requirement and can be accommodated in either centralized or decentralized design.	With respect to the ATS design, either system could provide the required power and transients although slightly more power at the beginning of life could be made available to the loads by use of switching regulators in either configuration.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	be taken into consideration. Most spacecraft depend on the battery to provide this transient capability and the system which permits the battery to come on the line most effectively is a consideration. The ability of the batteries to supply fault currents in an attempt to burn out fuses or faults must also be considered. In the case of the ATS spacecraft design, the battery comes on the line almost instantaneously through the battery controller, providing peak currents and fault burnthrough currents. The decentralized system described later also permits easy access to the battery for transient currents although design for high current transient is more complex in the centralized than the decentralized design.		
d) Electromagnetic Interference and Ripple	EMI is an important factor in both power system design and the centralization decision and usually represents a major unknown in spacecraft systems design. The EMI control plan can range from overstringent to insufficient. Many	This factor can be a key factor in the centralization decision.	With respect to the ATS spacecraft, it is expected that the centralized switching converters as concepted would not adversely affect

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	spacecraft design approaches tend to set rigid MIL spec type levels for subsystems forcing the subsystem hardware to include EMI filters, feedthrough, capacitors and shielded enclosures. While this is a safe plan, it can easily lead to excessive weight, cost, and complexity. On the other hand, too little attention to this problem has often led to interminable debugging and brute force filtering incorporated during spacecraft test. A reasonable test program imposed during subsystem development can be used to predict compatibility of subsystems during spacecraft integration and test. Measurements of susceptibility and conducted and radiated noise generation of each control assembly can be accomplished during development. A 6 dB spread between the maximum noise generated at a particular frequency by all spacecraft assemblies and the susceptibility of other assemblies is a typical safety margin. However, interaction with non-spacecraft equipment at the launch site may require special consideration. Tests run to determine the above described		the loads and that ripple suppression and interaction between loads can be adequately controlled by small L-C filters.

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	margin must be set up with full understanding of the spacecraft distribution system so as to provide realistic source and harness impedances and load terminations. One of the desirable features of a decentralized system such as the ATS is the degree of isolation provided by the remotely located regulators. The remote regulators act as a filter for incoming ripple from low frequencies to the high kilohertz region depending on design. Capacitors normally required by these regulators for stability also assist in minimizing ripple feedback. Isolation required in a centralized system is provided by L-C filters. While ripple suppression and load isolation can normally be provided in this manner, it is probably never equivalent to the active filtering provided by the regulators. The EMI normally generated by the switching regulator in the centralized design is not expected to cause problems with most spacecraft loads. This is based on experience on several spacecraft including the ATS.		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	For the ATS-E, it was necessary to provide a switching converter between the bus and the loads due to the heat pipe experiment. The heat pipe experiment, included after the panel had been designed, lowered the panel temperature and raised the voltage. The switching converter was added to act as a dc transformer to match the desired load voltage of 27 volts, thereby increasing load power. The addition of this switching converter imposed no EMI problems on the spacecraft. In fact, it actually lowered the low frequency bus ripple due to the large filtering required by the converter design.		
e) On/Off Requirement and Overload Protection	These factors are an important consideration in the power systems design. All spacecraft systems design must consider the means by which loads will be connected and disconnected from the bus and how loads are to be isolated in the event of a fault. The decentralized system used on ATS lends itself to incorporation of on/off control and overload protection since these functions can be built into the remote regulators, generally with	On/off control and overcurrent protection are not particularly significant to the centralization decision.	On the ATS spacecraft, both of these functions are incorporated in the remote regulator. No difficulty is seen, however, in providing equivalent performance in a centralized design.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	little or no additional power loss. Overload protection provided in this manner can be made precise by increasing the series drop or circuit complexity. However, overload protection provided in this manner is never complete. That is, there is usually some failure which will still cause high fault current. The ATS spacecraft uses the remote regulators for on/off and overload protection. Overload protection level is a function of transistor gain and is therefore not precise. In the concepted centralized system, on/off control is provided by commandable relays and fault protection by fuses.		
f) Number of Loads	The number of loads can be an important factor in the centralization decision. With only a few spacecraft loads, remote regulators are advantageous because the number of parts and the cost and reliability of the simpler remote regulators compare favorably with the more complex-redundant centralized concept. As the number of loads increases and more and more remote	The number of loads is a key factor in the centralization decision.	With respect to the ATS spacecraft, there are a sufficient number of loads to warrant serious consideration of a centralized system.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	regulators are required, the tradeoff with respect to parts count and reliability eventually favors the centralized system. This tradeoff, of course, is independent of the other factors listed in this study.		
2) Load Sequencing	Load sequencing determines maximum power required from the solar panel, maximum energy storage required by the battery, and any high current modes of operation. The latter factor can be critical, particularly in the area of squib firing when more than one squib may be required to fire at one time. Peak current capability at that time is important to mission success, and the power system must be designed to provide this high pulse load. In providing this type of load, it is sometimes necessary to run a special bus directly from the battery or "battery bus" to assure availability of peak load currents. If this is not done and all loads are taken from the bus, the battery discharge path must be analyzed with respect to diode and transistor peak power and currents and the associated voltage drops and other inherent current limiting features.	The load sequencing may affect the centralization decision, particularly when battery charge is appreciable.	The ATS spacecraft charge power is small and does not have to consider maximum power tracking. The ATS load sequencing requirement can be met using either centralized or decentralized design. Both systems result in a simple straightforward design. The centralized design could provide more power at the beginning of life.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	Prelaunch, launch, and transfer orbit power requirements are determined by which loads must be on and for how long. The time the spacecraft operates on the battery and the availability of the solar panel and its orientation together with the sequencing of the load play an important part in finalizing the power system design. An example of load sequencing taken from another program is shown in the histogram of Figure 7-5, based on the power budget shown in Table 7-1 which shows the power required during various modes of operation. The power consumption in Figure 7-5 is shown as a function of the number of orbits throughout the mission life of the spacecraft. The power system design must then be matched to this load profile. The battery must sustain the entire spacecraft load until solar panel power is available. The solar panel must be sized to provide the peak power requirement which, in this case, occurs during Mode 9 (battery charge). A secondary peak power occurs during Mode 3 (Experiment 2, erect and stow). In this particular		

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	satellite, this mode occurs before full panel power is available and must be supplied by the battery. The above description is intended to provide an example of a typical satellite load histogram and its importance as a power system design factor. The high charge power shown in this histogram would greatly influence the power system configuration. For effective design, it might be necessary to use a maximum power tracker for battery charging. The centralized system connects the battery directly to tapped sections of the panel for battery charging. That concept, though advantageous for ATS, might not be compatible with "maximum power tracker" design. The final orbit is subject to seasonal and daily variations. Figure 7-6 shows the load requirements of a synchronous communications satellite as a function of season. The power requirement is steady except during eclipse season when battery charging is required. For this requirement, battery charging, unlike the previous histogram described,		

TABLE 7-1. POWER BUDGET

Power at 28 Volts, watts

Subsystem	Mode 1 (Launch)	Mode 2 (After Spinup)	Mode 3 Experiment 2 (Erect and Stow)	Mode 4 (Mapping)	Mode 5 (Playback)	Mode 6 Experiment 3 (Operation)	Mode 7 Experiment 2 (Telemetry)	Mode 8 (Eclipse Operation)	Mode 9 (Battery Charge)	Mode 10 Experiment 1 (Retract)	Mode 11 Experiment 4 (Operation)	Mode 12 Experiment 2 (Redeploy)	Mode 13 (Failure Eclipse)
Experiment group													
Experiment 1													
Experiment 2													
Erect and stow			500.0										
Cooler	500.0	500.0	500.0	500.0	500.0	500.0	500.0	500.0	500.0	500.0	500.0	500.0	
Telemetry			74.0	74.0			74.0					74.0	
Sensor				26.0								26.0	
Pump	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	100.0	
Experiment 3						37.0		37.0	37.0				37.0
Experiment 4											8.0		
Communication and data handling													
T&C digital	23.0	23.0	23.0	23.0	23.0	23.0	23.0	23.0	23.0	23.0	23.0	23.0	23.0
Receivers	14.0	14.0	14.0	14.0	14.0	14.0	14.0	14.0	14.0	14.0	14.0	14.0	14.0
Transmitter	19.0	19.0	19.0	19.0	19.0	19.0	19.0			19.0	19.0		
Data record				8.0		8.0		8.0	8.0			8.0	8.0
Data playback					18.0								
Despin		24.0	24.0	24.0	24.0	24.0	24.0	24.0	24.0	24.0	24.0	24.0	24.0
Edge tracker			20.0	20.0								20.0	
Electrical power													
Sensors, etc.	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0	12.0
Experiment 3 Conditioner						8.0		8.0	8.0				8.0
Heaters	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
Battery charge									728.0				
Total	678.0	702.0	1296.0	830.0	720.0	755.0	776.0	736.0	1464.0	661.0	710.0	811.0	136.0

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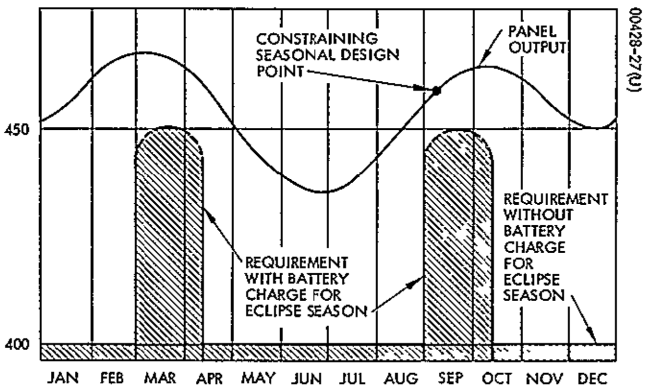


Figure 7-6. Load Requirements as Function of Season for Synchronous Satellite

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	is only a small percentage of the total power. This is because in synchronous orbit, the eclipse time is only 1.2 hours maximum out of every 24 hours, permitting recharge at a C/10 - C/15 rate. Superimposed on this figure is the solar panel output versus season showing the constraining seasonal design point which indicates the time when the smallest margin between available and power requirement occurs. Four of the five ATS spacecraft are synchronous and have seasonal variation as described above.		
3) Solar Panel Sizing	The load requirements and load sequencing previously discussed determine the panel power requirements. In order to size the panel, it can be seen that many of the other factors shown in Figure 7-1 must be considered. The configuration of the panel is usually determined by other than power system considerations. Overall spacecraft configuration, method of stabilization, launch vehicle constraints, etc. usually determine the configuration of the	Solar panel sizing can affect the centralization decision.	The ATS panel sizing would not be changed appreciably by using a centralized design. However, by using the centralized switching regulator design described in this report, more power at the beginning of life could be available to the experiments.

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	panels. Cost/size and weight trade-off of various panel configurations during preliminary design studies are used to assist in determining the overall spacecraft configuration. However, at the time the power system designer is faced with sizing the panel, the configuration of the panel has usually been narrowed down to a baseline configuration with only minor additional changes permitted. Once the baseline configuration has been established, sizing of the panel proceeds on the basis of optimizing the power system and the panel layout. The panel, battery, battery charge configuration, and distribution system must be considered as an integrated system in order to optimize panel sizing. In the case of the ATS, cylindrical panels were determined to have the least cost and most commonality for the five spacecraft. Oriented panels were considered to be in possible conflict with the gravity gradient experiment. Cylindrical panels were determined to be less costly to manufacture than other body		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	mounted arrangements such as bolted flat sections. The charge configuration on ATS was selected as a small boost array mounted in series with the main array and permanently connected to each battery. The distribution system was selected as decentralized with remote regulators. With the above decisions made; the sizing of the panel could then proceed based on the load profile previously determined. An important input to the solar panel sizing is the panel thermal prediction since power is a function of temperature. Spacecraft life requirements, attitude, radiation environment, potential contaminating sources, shadowing; harness drops, and relability are all significant factors in solar panel sizing. If the spacecraft configuration dictates the use of oriented panels, tradeoffs of degrees of orientation and flexible versus rigid panels are required.		

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	The sizing of the solar panel is usually a major factor in the overall spacecraft size and is sometimes critical in determining the booster to be used. Power system design to minimize the solar panel size can be greatly influenced by the battery charge and distribution system. Use of optimum charge trackers can minimize the size of the panel and influence the centralization decision. Centralized regulation can also influence the size of the panel, particularly with respect to beginning of life and end of life power.		
4) Energy Storage Sizing	Energy storage sizing is also an important power system factor and is influenced by most of the factors which influence solar panel sizing, although the energy storage system is not as critical to the overall spacecraft configuration as is the solar panel. On the other hand, the method of charge and discharge is a greater factor in the centralization decision than is the panel sizing. This is because the distribution system and the charge and discharge control	Energy storage sizing is a key factor in the centralization decision.	With respect to the ATS spacecraft, the charge current is small and full eclipse power is not provided. The energy storage sizing has not been changed when going from the present ATS power system to a centralized system even though the centralized

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	together comprise the power subsystem electronics and it is the total electronics that must be compared to determine the advantages of a centralized design. The battery system concepted for the ATS centralized design requires no charge or discharge electronics which is an important advantage to that approach. If the orbit requires substantial battery charge current from the panel, then the energy storage system has a major effect on the power systems design. A centralized design of the type concepted in this study can use the spacecraft battery more effectively than the present ATS spacecraft design. This is due to two separate categories of improvement. The first is by elimination of the drop in the discharge control series transistor and the drive power necessary for it. In the present ATS system, battery power is connected to each spacecraft load through two regulators, the battery		design utilizes the battery more effectively.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	discharge control and the remote regulator. In the centralized design, the battery load interface is through the single centralized regulator. The second category of improved use of the battery is that the switching regulator action of the centralized regulator permits operating at a lower depth of discharge since lower current is drawn from the battery during its higher voltage periods of operation. This can be seen by referring to Figure 4-3. The latter advantage would disappear if the ATS discharge control used a switching regulator, which it does not. The energy storage sizing is a particularly major influence in power system design for orbits where the discharge and charge time are in the 30 and 60 minute time periods such as found in low earth orbits. In this type of orbit, high charge currents are required, greatly influencing solar panel size and charge methods. Efficient use of the solar panel by such means as power tracking and overcharge		

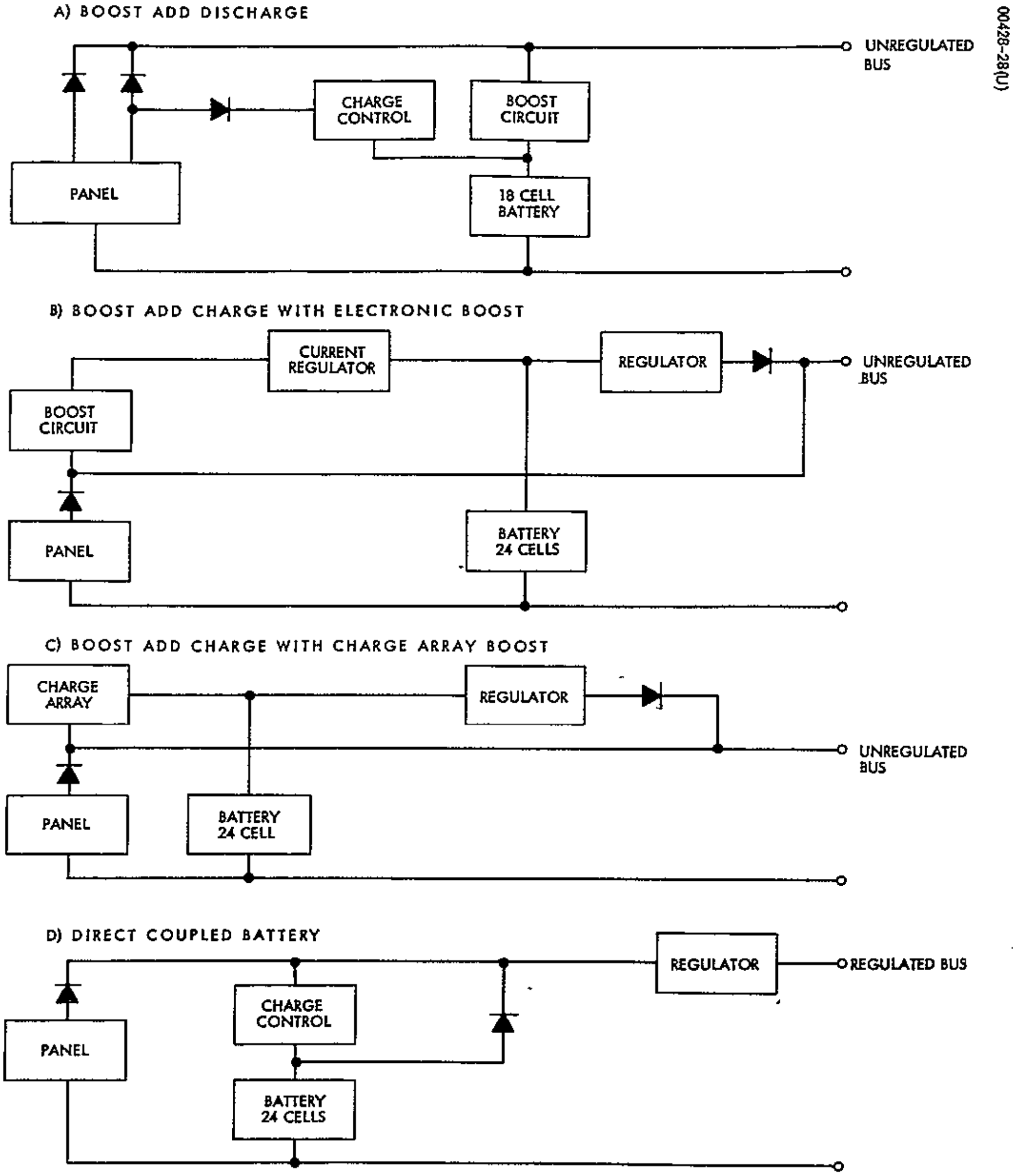


Figure 7-7. Basic Power Subsystems

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	control to avoid excessive temperatures must be considered.		
5) Charge Configuration	Many systems have been concepted to couple the battery to the panel for charging and to the bus for discharge. A few of them are shown in Figure 7-7. All of the systems can be used in either centralized or decentralized systems. However, the direct coupled battery (Figure 7-7d) lends itself most easily to a centralized system since it has the added advantage of not requiring a discharge control circuit and can therefore utilize the battery most effectively. In this system the solar panel is designed with sufficient voltage to permit charging of the battery without additional boost and under all conditions of solar panel operations the solar panel voltage is high enough to prevent battery discharge. During load transients or in eclipse, the panel drops and the battery discharges. The wide swing in voltage between maximum panel voltage and minimum battery discharge voltage (probably 2:1 minimum) requires a centralized regulator for reasonable system operation. Remote regulators could	Charge configuration is a key factor in the centralization decision.	In the case of the ATS spacecraft use of a centralized system would require a change in the charge/discharge configuration.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	also be used if all loads were coupled to the bus through them. The charge configuration is an important factor in power system design and in the centralization decision. The basic ATS system uses a boost add charge system with charge arrays (Figure 7-7c) and a series dissipative discharge control. A competing centralized system would use a high voltage panel with a fraction of the arrays ORed and paralleled to provide a charge bus (Figure 7-7d). No battery discharge control would be required as the batteries would be directly coupled to the bus.		
6) Distribution System	The distribution systems used is an important factor in power system design and is the primary subject of this study. Once the load requirements, load sequencing, and other spacecraft constraints are established, the power system design becomes a continuous iteration of solar panel	The distribution system is the primary subject of this study.	The ATS spacecraft was a completely decentralized bus system with limiters and battery discharge controls determining the upper and lower bus limits respectively. This study provided

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	and energy storage sizing, charge configuration, and distribution system. Tradeoffs of the various systems are continued until a final decision is reached. The distribution systems considered are decentralized and centralized regulation, centralized and decentralized dc to dc conversion, ac distribution, and various hybrid combinations. The ATS distribution system consists of two unregulated buses which can be interconnected via a paralleling relay on command. The upper limit of the bus is clamped by voltage limiters and the lower limit by battery discharge. All downstream regulation and conversion is accomplished by remote regulators and/or converter. A separate battery bus is run for pyrotechnic devices and other assemblies requiring direct access to the battery terminals.		detailed tradeoffs of the present system versus other systems with varying degrees of centralization.

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
OTHER FACTORS IN POWER SUBSYSTEM DESIGN			
1) Spacecraft Configuration	Spacecraft configuration is a major factor in the power subsystem design since it determines the constraints for the solar panel design and also the weight distribution and thermal environment of the battery and electronics. Spacecraft configuration is arrived at from a complex tradeoff of all subsystems and mission requirements. Once arrived at, it usually dictates the panel arrangement with respect to shape, dimensions, type of orientation, etc. Mounting of the spacecraft batteries is difficult. The batteries usually represent one of the heavier spacecraft components and also must operate within tighter temperature extremes than other spacecraft assemblies. In order to accommodate all the battery requirements, it is sometimes necessary to break the battery up into multiple packs. The number of battery packs and their location is a strong function of the spacecraft configuration and	Spacecraft configuration can influence the centralization decision if dissipation must be minimized. Also if spacecraft configuration and power tradeoffs result in the use of a rotary transformer, then a centralized distribution system would be indicated.	With respect to the ATS design, the spacecraft configuration can accommodate either a centralized or decentralized system equally.

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	usually is arrived at by a tradeoff of temperature profile, thermal control systems, weight distribution, and ease of mounting and replacement. Electronics mounting requirements are usually not too restrictive and suitable location providing acceptable structural, thermal, and electrical interface can usually be found without difficulty, particularly if dissipation is not excessive. Mounting of electronic equipment with significant dissipation can pose a problem to the spacecraft configuration. Some aspects of this will be discussed in the section of thermal control. The spacecraft configuration is important in determining the harness layout and associated voltage drops. This is even more pronounced if the spacecraft configuration requires the use of slip rings either for access to oriented panels or across other spinning interfaces. Harness drops are sometimes overlooked in preliminary design of power subsystem resulting in changes in design later in the program.		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	Spacecraft configuration affects harness layout and voltage drops and thus affects power subsystem design. The desire to eliminate slip rings for spinning interfaces as mentioned above also can affect the centralization decision. Use of rotary transformers as a substitute to slip rings has been considered and has been the subject of some developmental effort. If rotary transformers did prove advantageous, they might result in use of a centralized ac distribution system. The ATS spacecraft configuration utilizes cylindrical solar panels. Each battery is divided into three 6 cell packs and one 4 cell pack which is integrally mounted with the battery controller. The other electronic boxes are distributed around the spacecraft where space is available and harness layout can be accommodated. Distribution of the battery in several packs can cause problems if temperature Δs exist between the packs due to the marked effect of temperature on		

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	charge efficiency. The spacecraft configuration does influence the ability to distribute the battery in many packs for this reason. On the ATS the discharge controls and the limiters have high dissipation. Spacecraft configuration affects the thermal behavior of these boxes.		
2) Stabilization Method	The method of stabilization is an integral part of the spacecraft configuration. A spin stabilized spacecraft usually requires a body mounted solar panel design. A three axis stabilized method would require an oriented panel. Cells could be body-mounted but would more frequently be mounted on a separate oriented panel. Consideration of a separate panel for a spin stabilized spacecraft is not ruled out but tradeoffs have usually not indicated separate oriented panels to be advantageous. Gravity gradient stabilized spacecraft as concepted for ATS did not use oriented panels because of the potential interference with the gravity gradient experiment.	The stabilization method can affect the centralization decision due to the power variation of the panel.	The ATS spacecraft uses all cylindrical panels. The gravity gradient stabilized spacecraft and the spin stabilized spacecraft utilize similar power system configurations. The method of stabilization does not affect the centralization decision. The power output of the gravity gradient solar panel is substantially less than that of the spinning satellite due to the marked temperature difference.

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	The location of the jets is always associated with panel design both because of the potential shadowing effect and the potential high temperature associated with jet firing. If the stabilization method dictates the use of an oriented panel, the power performance of the panel might be sufficiently different from that of body-mounted panel in its effect on the centralization decision. The ATS uses two stabilization systems: one spinning stabilization, the other gravity gradient stabilization. The most pronounced effect of these two types of stabilization on the power system is the temperature of the solar panel. In the spinning case, the panel temperature is low due to the spinning action turning the panel alternately towards the sun and space. In the gravity gradient stabilized case, the panel faces the sun continuously (one revolution per 24 hours) and runs at a considerably higher temperature (except for the ATSE due to the heat pipe experiment).		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
3) Thermal Control	Thermal control for a spacecraft can use the power system as part of the thermal control system. One example is the TACSAT spacecraft which uses the power system in two distinct ways. First, a switch was provided to open one of the solar panels at a time when the communication system was disconnected. Removing a panel during this mode of operation reduced internal spacecraft dissipation. Second, the communication amplifiers and the power system voltage limiters were interconnected in order to control local hot spots. The limiters were fabricated with commandable set points. When an amplifier was turned on, the same command raised the set point of the adjacent limiter so that it would not turn on at the same time causing excessive local hot spots. Several ATS spacecraft also use limiters for thermal control but in a different manner. On several ATS spacecraft, the limiters are commanded on to provide a fixed load when needed to prevent excessively low temperature.	The thermal control system system and power subsystem components are frequently integrated to control spacecraft temperatures. A centralized design can appreciably change the dissipation of the spacecraft electronics boxes.	With respect to the ATS spacecraft high dissipating exist in the existing spacecraft electronics boxes. These would be reduced appreciably by using a centralized system. The effect on the battery and panel would not be significant.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	The thermal control design of the spacecraft is a major interface for the power subsystem, particularly in the solar panel design and battery battery temperature profile. Solar panel power is a function of solar panel temperature and seasonal variation and can have an important effect on panel sizing. Battery temperature profile is the key to battery performance and the thermal control system - battery package design and loaction are critical to cycle life and voltage performance. On the ATS spacecraft, battery charge current is low and the resulting battery temperatures are satisfactory Dissipation of the ATS voltage limiters is high Some effort was required to understand power division between limiters so as to prevent any one limiter from overheating. On some of the later		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	units, six limiters were used instead of four to further reduce dissipation of any single limiter. The dissipation of the discharge control must be understood A typical curve of discharge control temperature and dissipation is shown in Figure 7-8 As eclipse is entered, the case temperature of the battery controller rises due to the dissipation on the series transistor The peak temperature depends on the thermal mounting and the rate of spacecraft cooling in eclipse The centralized system described eliminates the limiters and discharge control Other power subsystem electronics have a low dissipation Also, the downstream regulators are eliminated, lowering dissipation on the loads. This would have a major influence on the thermal control system		

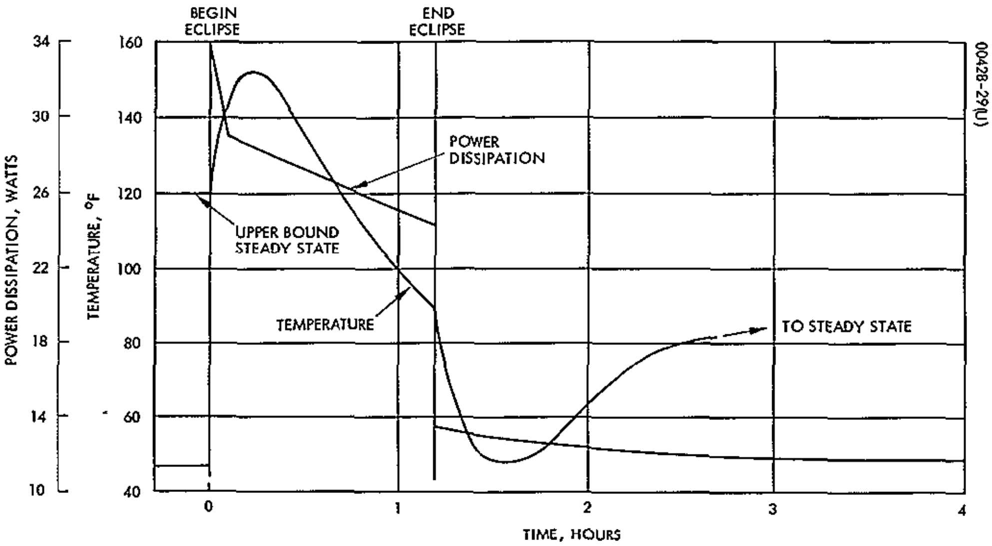


Figure 7-8. Battery Controller Discharge Pulse

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
4) Launch Vehicle	The launch vehicle determines the vibration, shock, and other environments that the power subsystem is subject to. It also dictates the spacecraft weight and volume constraint, which translates directly to the power subsystem. For the battery and electronics the launch environments imposed by the launch vehicle can usually be accommodated with normal design practices. It is usual to foam electronics assemblies to prevent vibration damage. Vibration environment for battery cells and battery packaging has not been unusually severe in most applications. The solar panel is most affected by the launch vehicle environments particularly when the solar panel is also a structurally loaded member. Tests on panels with combined environments simulating actual launch thermal shock and vibration conditions are needed to provide confidence of performance. The launch vehicle and desired orbit can influence battery electrical requirements in determining how long the spacecraft loads must	The launch vehicle determines the permissible weight and volume and many of the environments that the power system will be exposed to. It may also determine battery capacity. It does not usually affect the centralization decision.	With respect to the ATS spacecraft the launch vehicle would not influence the centralization decision.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	operate on battery power before solar panel power is available. Other important environments imposed by the launch vehicle are acoustic noise and fairing contimination. The former can be more severe than vibration in its effect on the panels. Fairing contamination is a factor in solar panel output. There is also a power interface with the launch vehicle and compatability with that is required. However, this does not usually become a major constraint. The ATS launch vehicles are the Atlas-Agena and Atlas-Centaur. The major environments are described in the requirements section.		
5) Launch and Orbit	The launch and orbit are key factors in power system design and in the centralization decision. The launch determines the time the spacecraft must run on battery power before the panel is illuminated. It also determines some of the	The launch plan can affect the energy storage system substantially in terms of total energy storage and peak power for squib firing and deployment.	The launch requirements for the ATS spacecraft did not affect the battery sizing but a separate battery bus was run for squib firing. The orbits of the ATS spacecraft (both

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	constraining environments including maximum panel temperatures. Many of the pyrotechnic devices are used during various phases of the launch and impart both shock and electrical pulse loads to the power system. Determination of the need for a separate battery bus usually results from analysis of the launch sequence. The launch also requires critical maneuvering by the spacecraft propulsion system, resulting in high temperatures in localized areas. The temperature and duration of these pulses is a function of the launch constraints. Battery sizing may be a direct function of the launch plan, particularly if the solar panel deployment is delayed for any significant time requiring greater stored energy during launch than in orbit. The final spacecraft orbit has a major effect on the power system, determining many critical factors such as seasonal effects, temperature extremes, light-to-dark time with	Critical temperature can also occur during launch. The final orbit affects the battery system and panel appreciably as it can drastically effect battery charge power requirements and the battery charge/discharge regime. The centralization decision is also a function of the spacecraft orbit.	synchronous and medium altitude) permit a simple power system without the need for overcharge control or high charge power. The number of eclipses per year is also not overly restrictive. The ATS orbits permit equivalent implementation of either a centralized or decentralized power system.

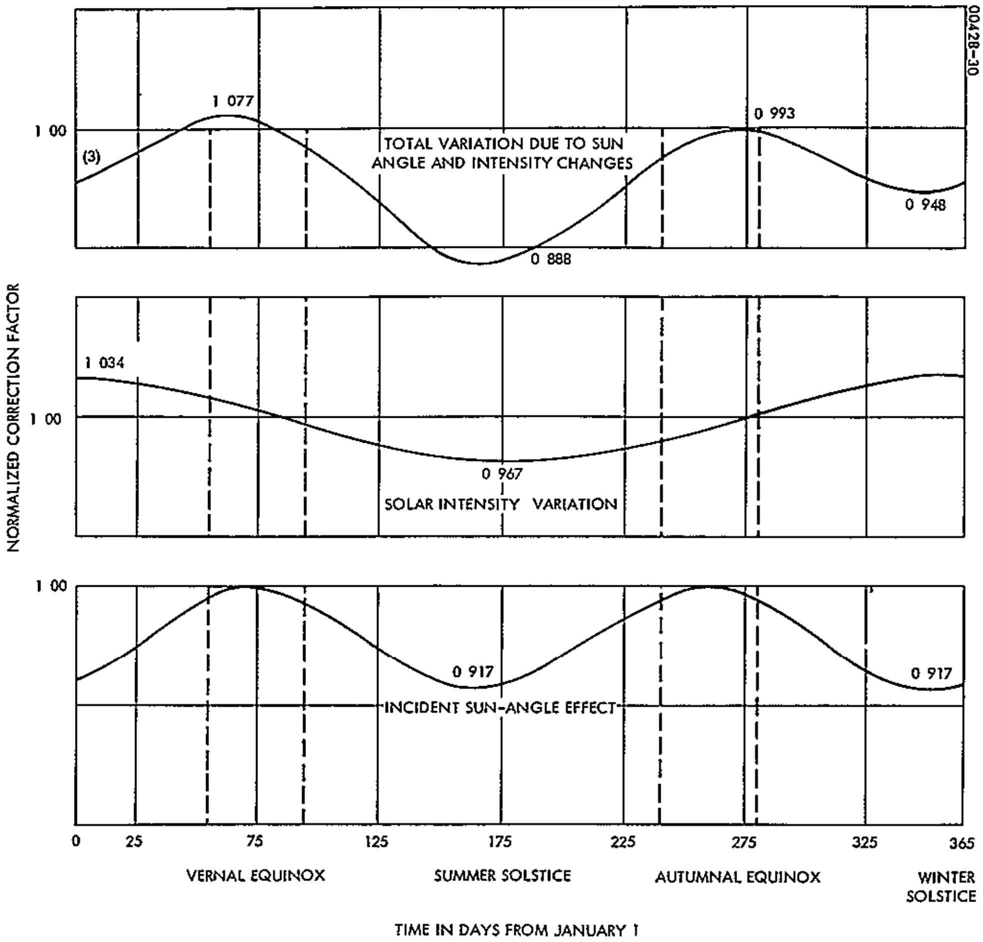


Figure 7-9. Solar Input Energy Variation for Typical Year

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	its pronounced effect on energy storage, depth of discharge, and charge power required. Figure 7-9 shows the seasonal effects of sun angle and intensity and Figure 7-10 shows the duration of eclipse seasons during the year on a synchronous satellite. In this type of orbit charge power is low and the number of eclipse cycles per year is low (90) permitting high depth-of-discharge without compromising life. Because of the low charge rates, overcharge control is not usually a requirement. The eclipse times of a contrasting orbit are shown in Figure 7-11. This 400 mile polar orbit introduces about 3600 eclipses per year, the longest approximately 36 minutes of duration with recharge time of approximately one hour. In this type or orbit recharge current is high. This orbit results in a completely different power system requiring overcharge control to prevent excessive battery temperature, lower battery depth of discharge because of the great number of cycles, and substantial additional solar panel area for battery charging.		

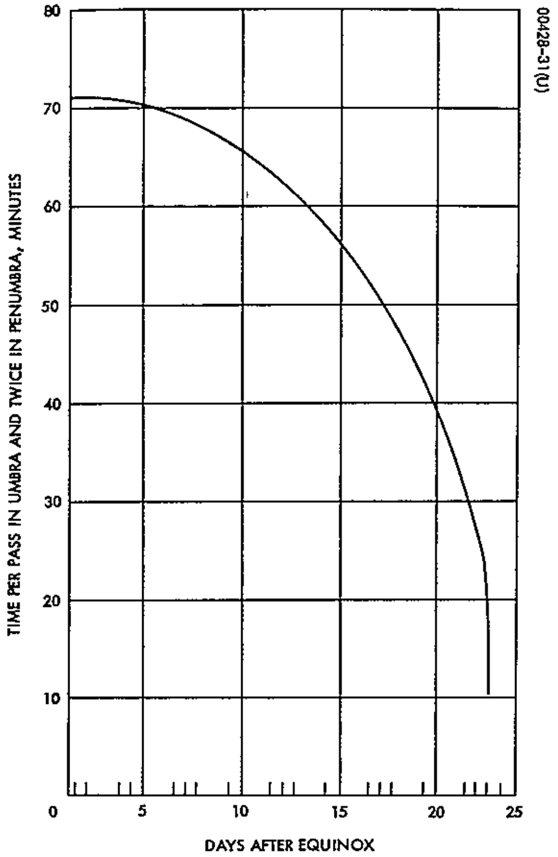


Figure 7-10. Days After Equinox Cycle Versus Time in Eclipse Synchronous Equatorial Orbit

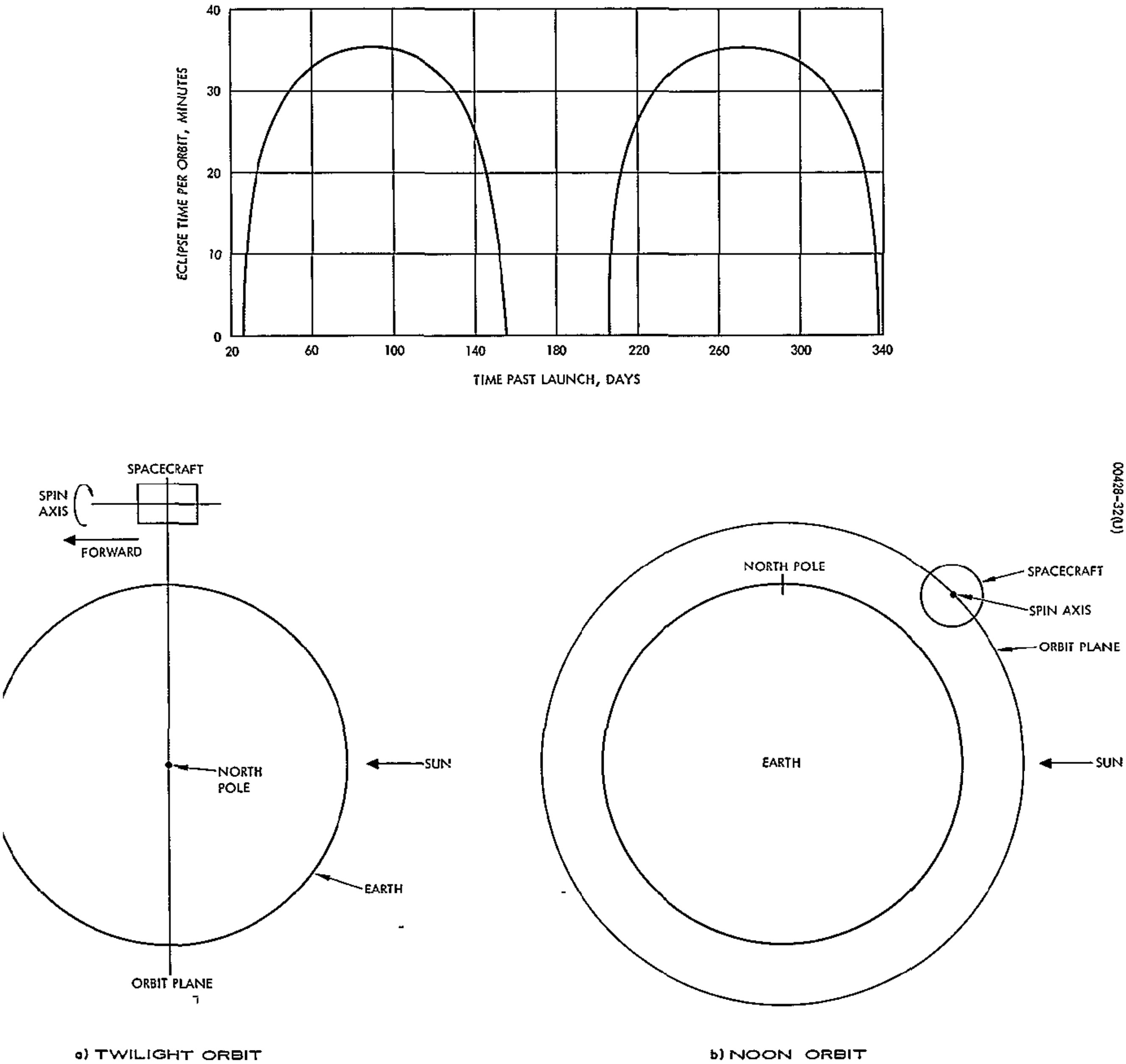


Figure 7-11. Eclipse Times Polar Orbit

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	The orbit thus affects the sizing of the power system. It also affects the centralization decision since the need for higher charging power and overcharge control affect all the system tradeoff. The ATS spacecraft has two orbits, one synchronous and one medium altitude. The medium altitude uses a higher charge current than the synchronous orbit (approximately 2 to 1) and operates the batteries at a lower depth-of-discharge in order to achieve the additional cycle required (1370 versus 90/yr). The remainder of the power system configuration for the medium design is similar to that of the synchronous altitude design.		
6) Cost and Reliability	Cost and reliability are always factors in any spacecraft power system design. Reliability is usually set up as a program requirement, and cost is one of the tradeoff factors to be considered in the power system design and centralization decision.	Reliability and cost are both important factors in the centralization decision.	For the ATS spacecraft better reliability is achieved in a centralized regulation system.

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	If two systems are assessed independently, it can turn out that the most reliable is also the least expensive. On the other hand, building additional reliability into a system usually raises the cost. The present ATS power system has several reliability features. Two buses are used with a commandable parallel connection to prevent losing the entire spacecraft bus. The parallel connection automatically opens if a fault develops across the bus system. The battery discharge controls are protected against excessive transistor dissipation prior to the parallel connection opening. Each load is isolated by its own current limited regulator. The centralized system concepted for this study has additional reliability features. The centralized regulators are redundant so that full capability is maintained with a regulator failure.		

Design Factor	Design Factor Discussion	Summary: Bearing on Centralization Decision	Summary: Bearing on ATS Decision
	A crowbar circuit protects against overvoltage in the event of a series transistor short in the switching regulator. Fuses are provided for overcurrent protection for each load. The reliability predicted for the centralized system concepted for this study is better than that of the present design.		
7) Radiation	Trapped, solar flares, and nuclear radiation environment are power system design factors. Trapped radiation affects electronic equipment to the extent of long range change in transistor gain and leakage characteristics and may also cause changes in other components such as precision resistors and diodes. However, design of electronic equipment can take these tolerance effects into consideration. The resulting designs are not unduly penalized by the effects of trapped radiation. Batteries are not considered to be affected by trapped radiation.	Trapped and solar radiation have considerable influence on solar panel design and a lesser influence on electronic design. The centralization decision is affected in using the beginning of life power effectively. Nuclear blast radiation has a major effect on panel electronics component, and circuit design and effects the centralization decision.	The effect of trapped and solar activity was taken into consideration in the ATS electronics and solar panel design except that exposed silicon around the coverglass caused unexpected degradation. A centralized ATS power subsystem would have more power at the start of life.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	The effect of trapped radiation on solar cells has been the subject of much study and testing. This extensive subject is beyond the scope of this study. However, as a power system design factor, it is noted that severe solar cell damage is expected to occur to an unprotected solar cell due to the radiation environment seen by the spacecraft. The panel prediction technique used is to reduce the expected environment analytically to 1 Mev equivalent electrons as a function of coverslide and cell parameters. Figure 7-12 shows the resulting effect of the 1 Mev fluence on cell parameters. The solar cell resistivity and thickness, coverglass thickness, and effectiveness in preventing exposed silicon are all factors which affect the panel weight, performance, and cost and which must be traded off in arriving at the final design. Nuclear radiation, if specified, can have a drastic effect on power system design in both the panel and electronics areas. Analysis and component and circuit test are		

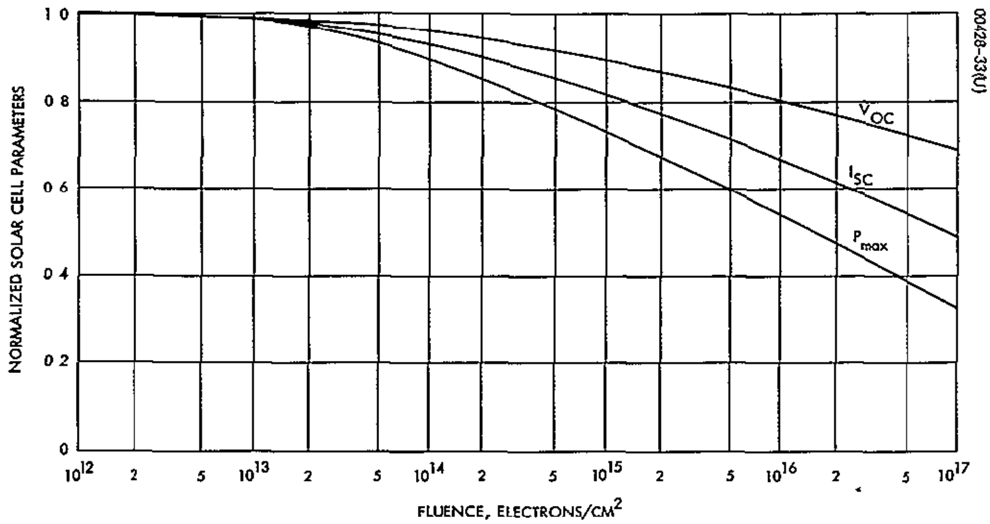


Figure 7-12. Normalized Solar Cell Parameters Versus Equivalent 1 Mev Electron Fluence

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	required to determine the total effect of any high level radiation environments. The ATS electronic design used a predicted transistor gain decrease of 2 to 1 and a leakage current increase of 10 to 1 for three years of operation. The panels used 12 mil 10 ohm-cm cells with 30 mil coverslides. No protection for coverslide gaps was provided initially. Anomolous panel performance led to development of a technique of applying a thin epoxy coating over the exposed silicon resulting from coverslide gaps. Zero gap type cell and cell-cover combinations have been used as another approach to this problem. With respect to the centralization decision, trapped and solar radiation effect expected degradation at end of life. The use of the extra power available at the beginning of life could influence the system configuration, Operation after exposure to nuclear radiation affects the type of components and circuits used for		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	the electronics and could also influence the centralization decision. The centralized system concepted for the ATS does provide more beginning of life power due to the expected panel degradation.		
8) Mission Objectives and Other Special Requirements	Mission objectives and special requirements must also be considered in power system design. A brief example of the effect of mission objectives is the TACSAT spacecraft solar panel switch. This switch was placed in series with a solar panel as a part of the thermal control system in order to increase reliability by lowering dissipation in certain operating modes. However, failure of this switch in the open position would disable half the spacecraft power and limit the primary mission objective of high power communications. The design approach to this switch was dictated by the mission objectives to fail short instead of open, being sure to provide sufficient power at the possible expense of slightly higher dissipation.	Mission objectives and special requirements must be considered in the centralization decision.	On the ATS spacecraft there are no mission objectives or special requirements that would affect the centralization decision.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	An example of a special requirement is magnetic fields. This factor can influence the power system design in several ways. One influence is in the use of magnetic materials such as Kovar and nickel or nickel plating. Nickel-cadmium batteries might not be satisfactory on this basis. Another is that the solar panel and harness arrangement can be laid out to minimize induced fields. Alternate solar cell strings can be laid out to carry currents in opposite directions and also if two panels are used, they can be laid out so that their fields cancel. Harness arrangements can be similar. Some ATS spacecraft used magnetometers to make magnetic measurements. They were mounted on booms to reduce the field in their immediate areas. Solar panel and harness layout had the objective of magnetic field reduction. Mission objectives and special requirements can have an effect on the centralization decision. However, it is not obvious that there are mission objectives or special		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	requirements not listed in other factors which would effect the ATS centralization decision.		
9) Interface Requirements	The power subsystem interfaces with the spacecraft load, harness, structure, thermal control, test equipment, launch vehicle, and telemetry and command subsystems. Interfaces with the load, structures, thermal control, and launch vehicle have been discussed previously. Interface with the harness and test equipment is usually straightforward. The interface with the command and telemetry subsystems is a function of the number of channels available and the signal levels and signal conditioning required. Voltage signals for telemetry are usually provided either through a high impedance divider with approximately 5 volt output or through a current limiting resistor with the divider in the telemetry subsystem. Current signals can be provided by magnetic type current to voltage transducers or by low level signals from shunts. If shunts are used, the amplifier can	Many of the interface factors have been summarized previously. Telemetry channels have minimum influence on the centralization decision but the available commands can have a larger influence.	The centralized ATS design would require 2 more command than the present design.

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	either be in the power subsystem or in the telemetry. It is usually more convenient for them to be in the telemetry due to the requirement for + and - supplies already present in the telemetry and not usually required in the power subsystem. Temperature signals are provided by temperature sensors. The signal conditioning is also usually located in the telemetry subsystem. Various status logic signals are also usually required. Only a limited number of commands are usually available to the power subsystem. The interface with the available commands is straightforward. Command interface must be designed to avoid activation by noise or open circuit. Commands can either be steady or pulse type. If pulse type, the power subsystem requires electronic or mechanical latching. The ATS power subsystem spacecraft uses few commands. Only the relay and the current control systems are commandable. The downstream regulators have on/off command capability.		

Design Factor	Design Factor Discussion	Summary	
		Bearing on Centralization Decision	Bearing on ATS Decision
	The ATS telemetry includes bus and battery voltage, bus currents and battery charge and discharge currents, and battery and solar panel temperatures. The remote regulators have on/off status signals. The telemetry available has little impact on the centralization decision. The commands available have a greater effect as certain systems can not be implemented due to the lack of commands.		

APPENDIX A

SCHEMATIC DIAGRAMS

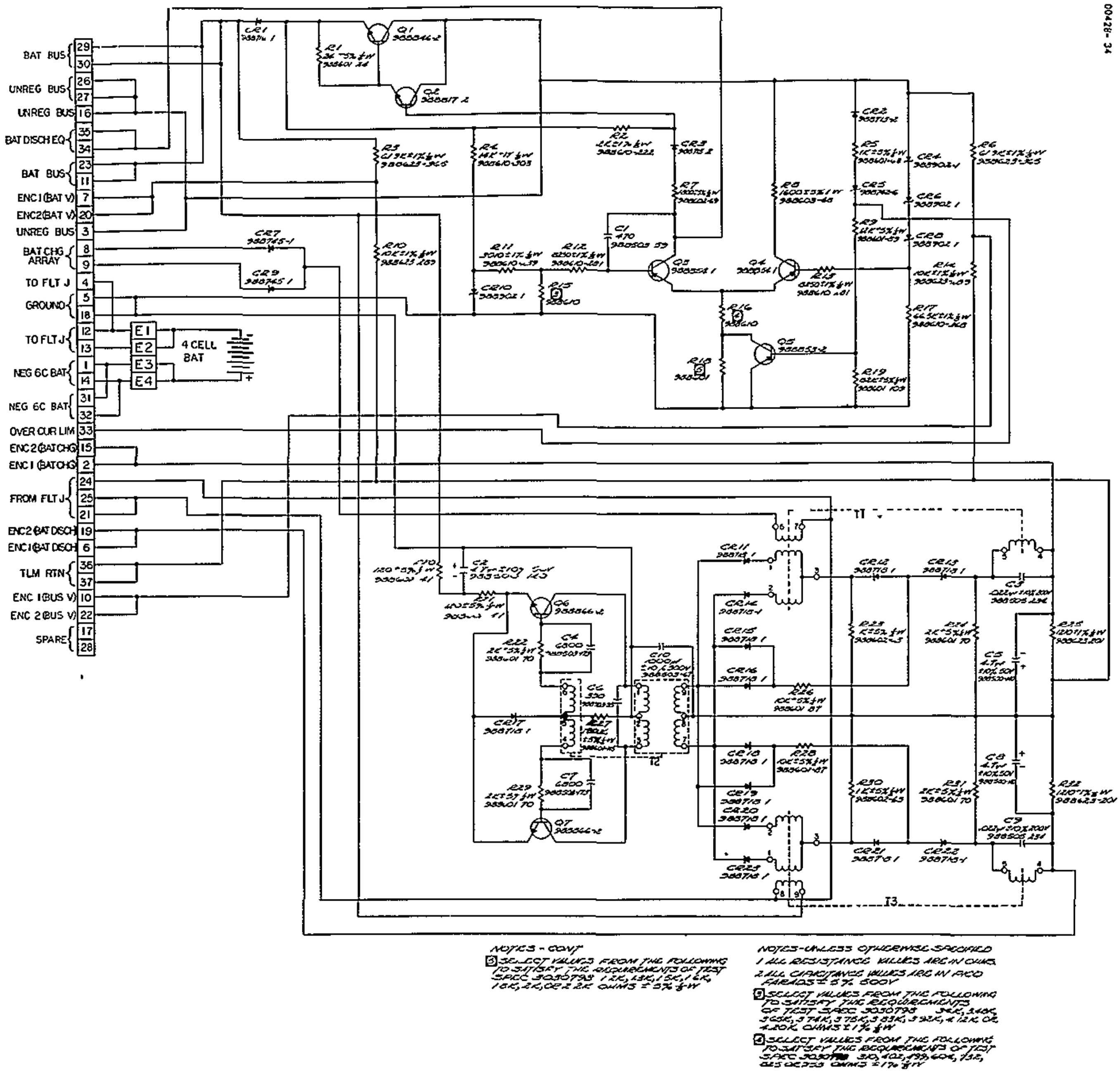
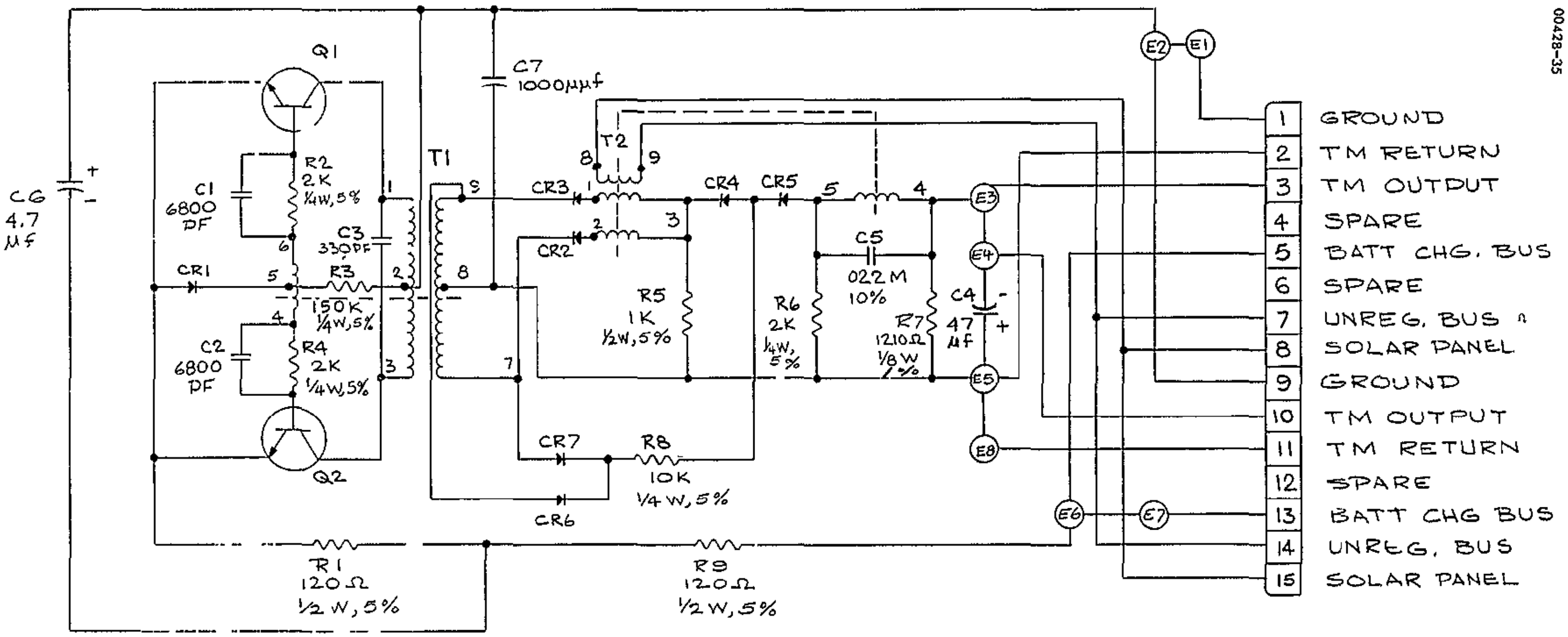


Figure A-1   Four-Cell Battery

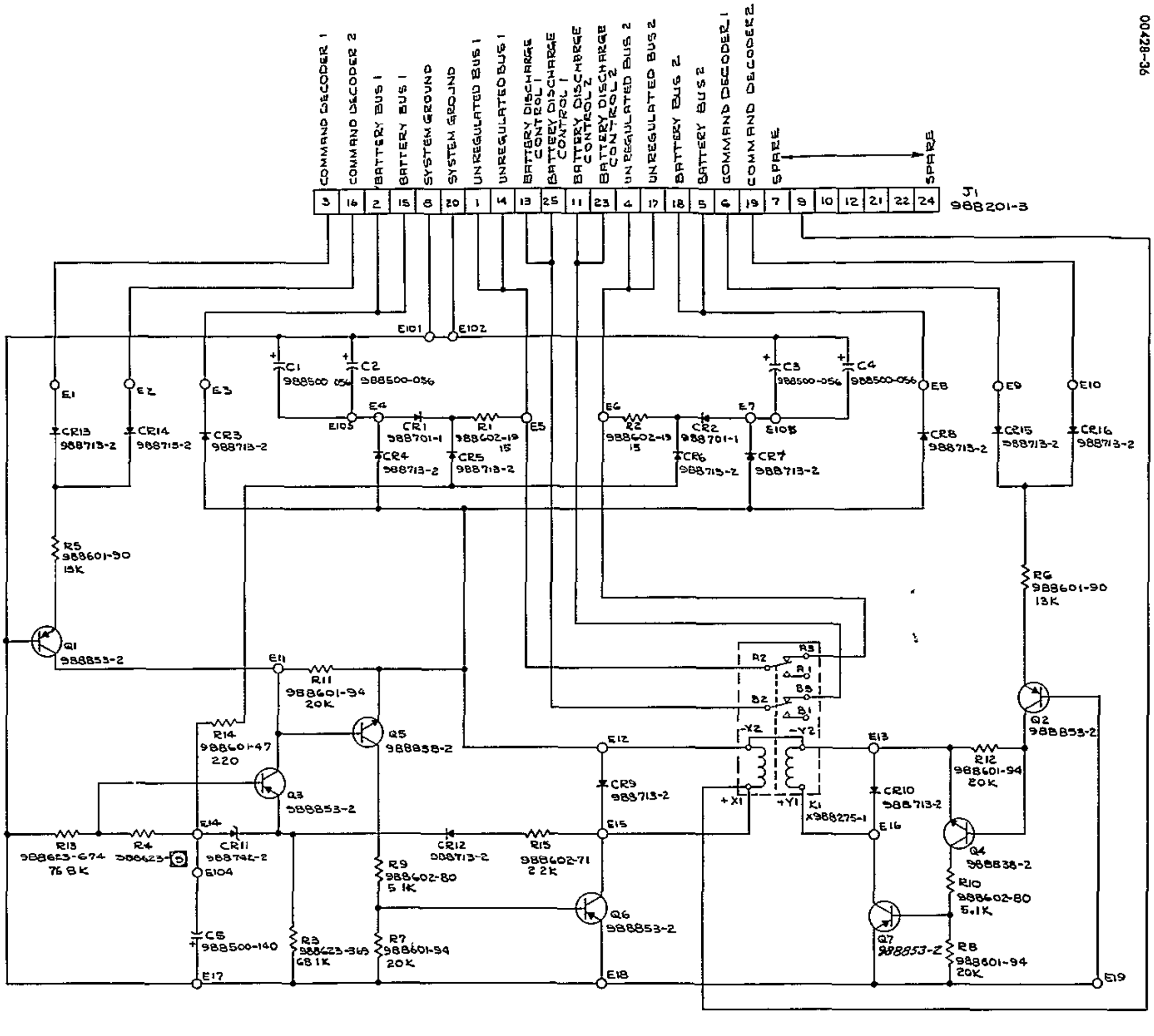


NOTES-UNLESS OTHERWISE SPECIFIED

1 FOR ASSY DWG SEE 475271-100

2 PARTIAL REFERENCE DESIGNATIONS ARE SHOWN FOR COMPLETE DESIGNATION PREFIX WITH UNIT NUMBER OR SUBASSEMBLY DESIGNATION

Figure A-2. ATS Current Sensor



NOTES-UNLESS OTHERWISE SPECIFIED

1 FOR ASSEMBLY DRAWING SEE 475272-100

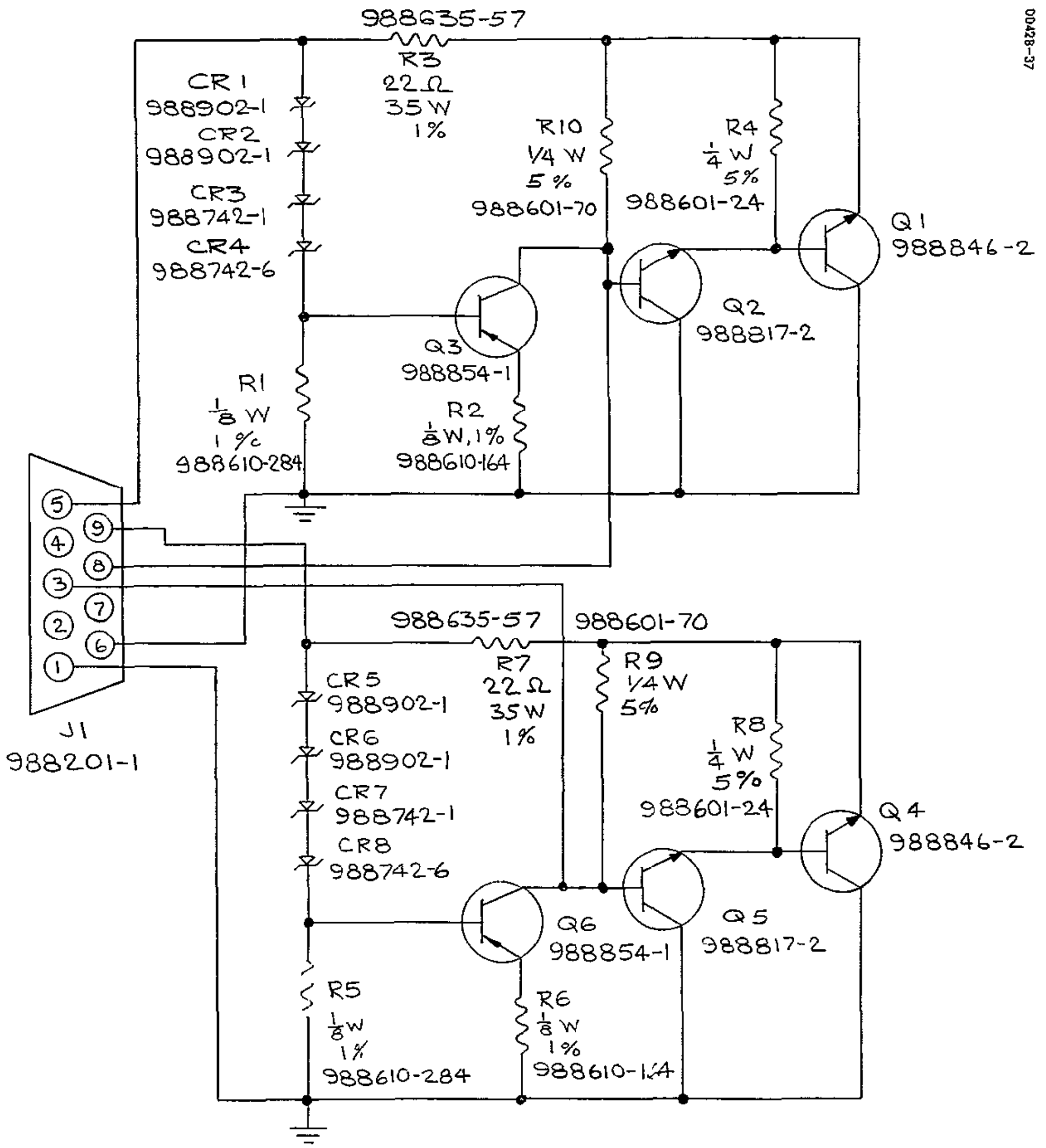
~~2. FOR WIRING DIAGRAM SEE DWG 475272-200~~

3 PARTIAL REFERENCE DESIGNATIONS ARE SHOWN. FOR COMPLETE DESIGNATION PREFIX WITH UNIT DESIGNATION

4 ALL 988XXX SERIES NUMBERS ARE VENDOR ITEMS - SEE SPEC CONTROL DRAWING

5 SELECT CORRECT DASH NUMBER FOR VALUE REQUIRED FROM 988623-338 THRU 988623-346

Figure A-3 Bus Relay Switch



NOTES - UNLESS OTHERWISE SPECIFIED

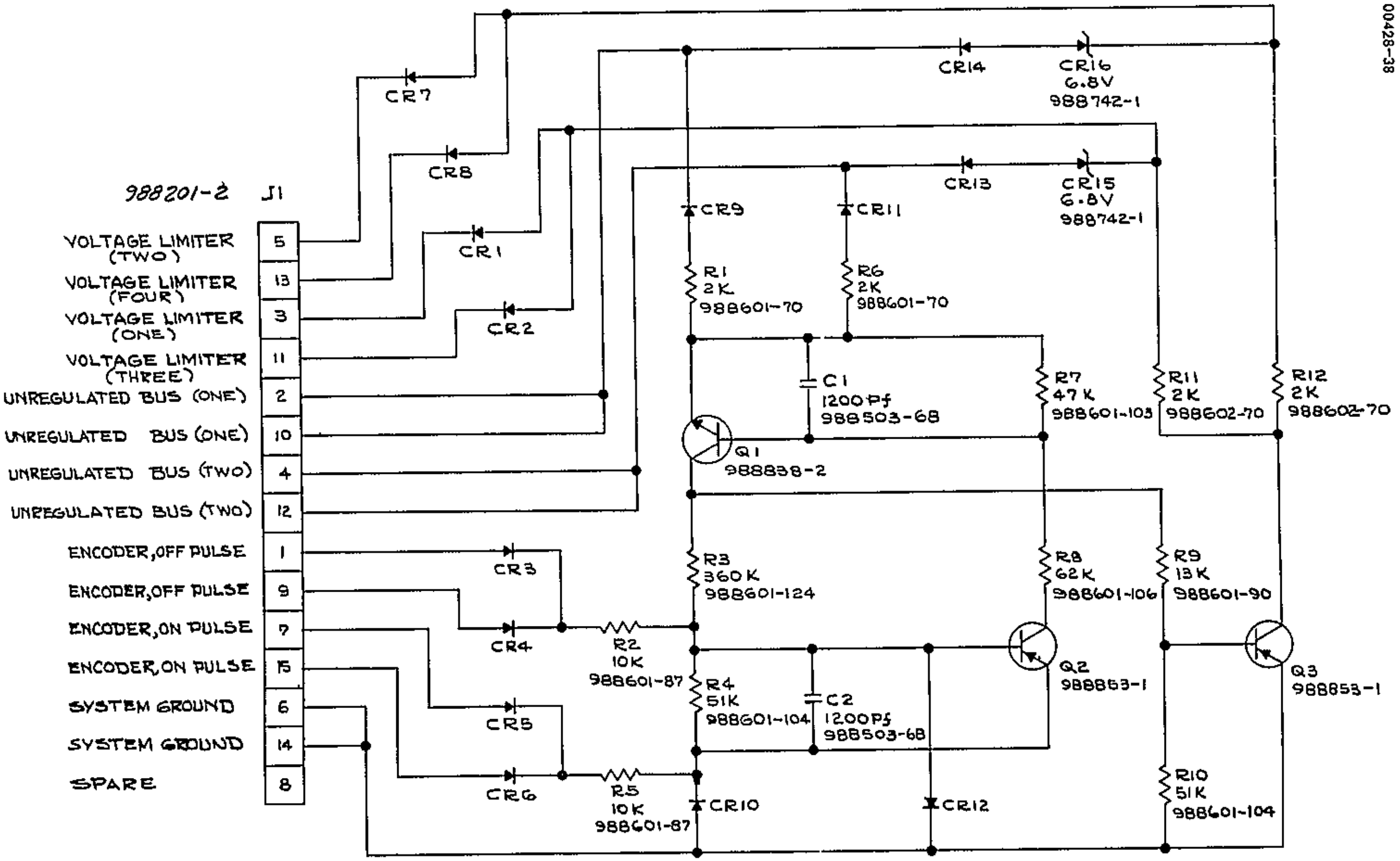
1 FOR ASSY DWG SEE 475270-100

2 FOR WIRING DIAGRAM SEE 475270-300

3 PARTIAL REFERENCE DESIGNATIONS ARE SHOWN FOR COMPLETE DESIGNATION PREFIX WITH UNIT NUMBER OR SUBASSEMBLY DESIGNATION

4 ALL 988XXX SERIES NUMBERS ARE VENDOR ITEMS, SEE SPECIFICATION CONTROL DRAWING

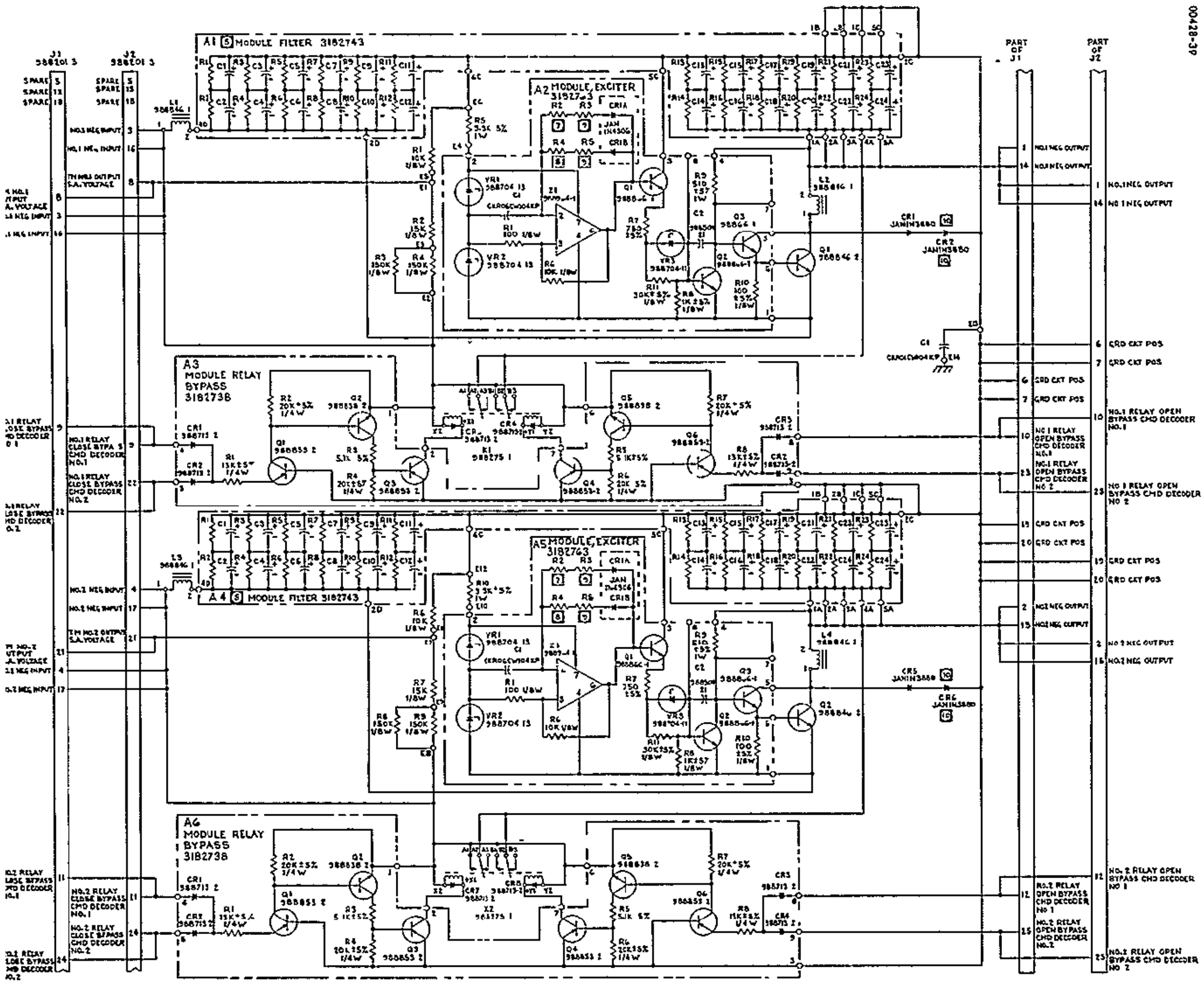
Figure A-4. Bus Voltage Limiter



NOTES—UNLESS OTHERWISE SPECIFIED

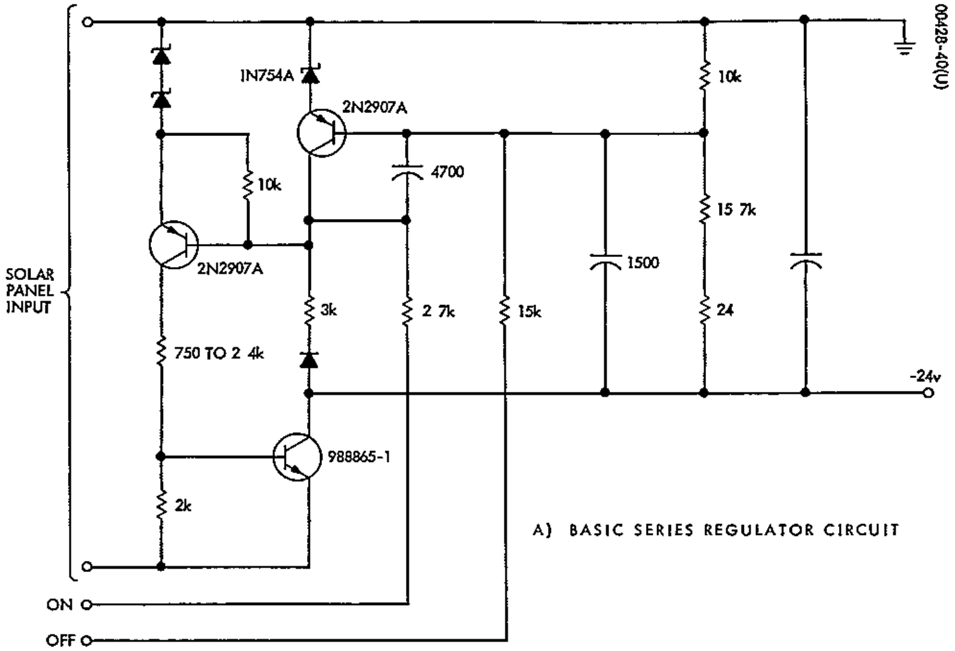
1. FOR ASSEMBLY DRAWING SEE 3030976.
2. FOR OUTLINE AND MOUNTING SEE 475273-500.
3. ALL RESISTANCE VALUES ARE IN OHMS.
4. ALL DIODES ARE 988718-1.
5. PARTIAL REFERENCE DESIGNATIONS ARE SHOWN. FOR COMPLETE DESIGNATION, PREFIX WITH UNIT NO OR SUBASSY DESIGNATION.
6. ALL 988XXX SERIES NUMBERS ARE VENDOR ITEMS, SEE SPEC CONTROL DRAWING.

Figure A-5. Current Control Unit



NOTES UNLESS OTHERWISE SPECIFIED
1 FOR ASSY SEE DRAWING 475274 100
2. ALL 5XXXXX SERIES NUMBERS ARE VENDOR ITEMS-SEE SPEC CONTROL DRAWING
3. RESISTANCE VALUES ARE IN OHMS ±1% 1/2W
4. CAPACITANCE VALUES ARE IN MF ±10% 100VDC
5 RESISTANCE VALUES ARE 100K ±5% 1/4W CAPACITANCE VALUES ARE 160 MF ±20% 50VDC
6 PARTIAL REFERENCE DESIGNATIONS ARE SHOWN, FOR COMPLETE REFERENCE DESIGNATION PREFIX WITH UNIT NO AND SUBASSEMBLY DESIGNATION
7 SELECT VALUES FROM THE FOLLOWING TO SATISFY THE REQUIREMENTS OF TEST SPECIFICATION 475274 600; 51 1K 54 9K, 57 6K 63 4K 78 7K ±1% 1/8W (RNR57C--); OR 59K 61 9K, 63 4K 64 9K, 68 1K 73 2K 75K ±1% 1W (RNR55C) OR 52.3K 53 6K 56 2K, 60.4K, 66 5K, 71 5K 76.8K, ±1% 1/8W (588623 X)
8 SELECT VALUES FROM THE FOLLOWING TO SATISFY THE REQUIREMENTS OF TEST SPECIFICATION 475274-600 15K, 15.4K, 15 8K, 16.2K, 16.5K, 16 9K, 17 4K, 17 8K, 18.2K 18.7K 19.1K, 19 6K 20K, 20.5K, 21K, 21.5K, 22.1K, 22.6K, 23.2K, 23.7K, 24.3K, 24.9K 25.5K, 26.1K, 26.7K, 27 4K 28K, 28.7K 29.4K 30.1K ±1% 1/8W (RNR57C)
9 SELECT VALUES FROM THE FOLLOWING TO SATISFY THE REQUIREMENTS OF TEST SPECIFICATION 475274 600; 511 562 604 619 681, 768, 825 ±1% 1/8W (RNR57C-) OR 523 536 576 590 634 649 698 715, 732, 750, 787 806, 845 866 887 909 931, 953 1K, ±1% 1W (RNR55C) OR 549 665 976 ±1% 1/8W (588623 X)
10 TESTED 1000 HRS AT 6A AT 100 C
11 ALL RESISTORS TOL ±1% ARE METAL FILM TOL ±5% ARE COMPOSITION
12 A2R1 & R6, A5R1 & R6 R1 THRU R4 & R6 THRU R9 ARE RNR57C

Figure A-6. Switching Converter



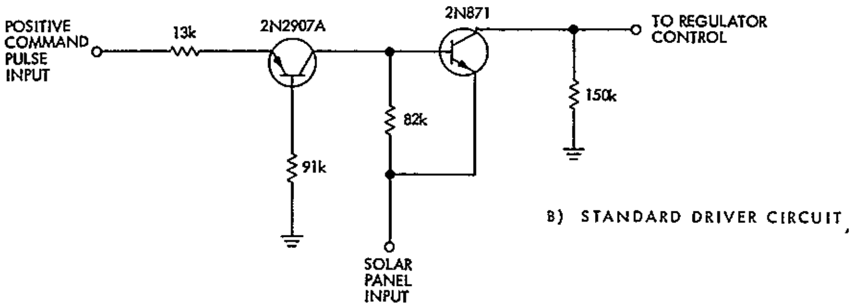
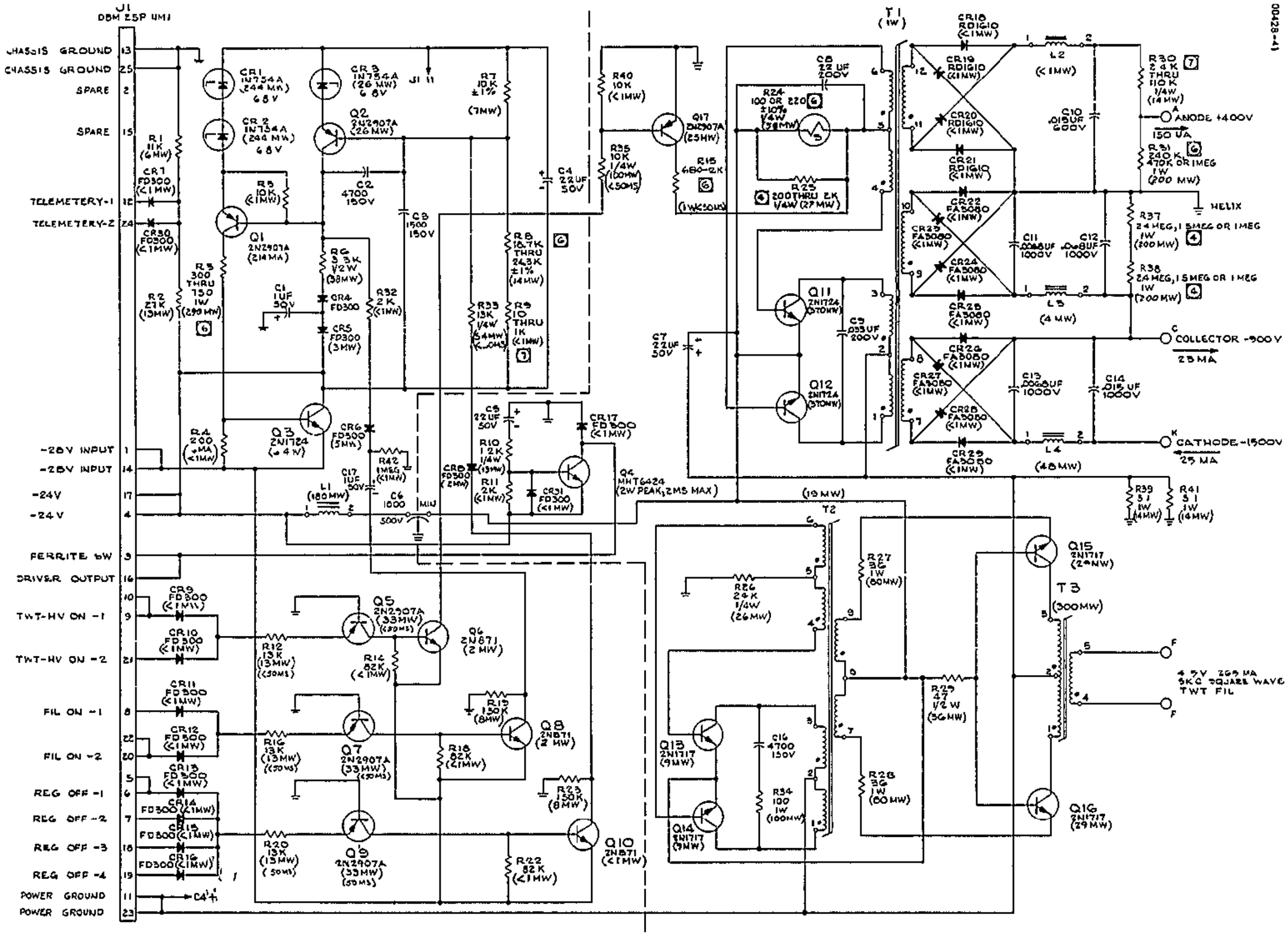


Figure A-7. Circuit Drawings



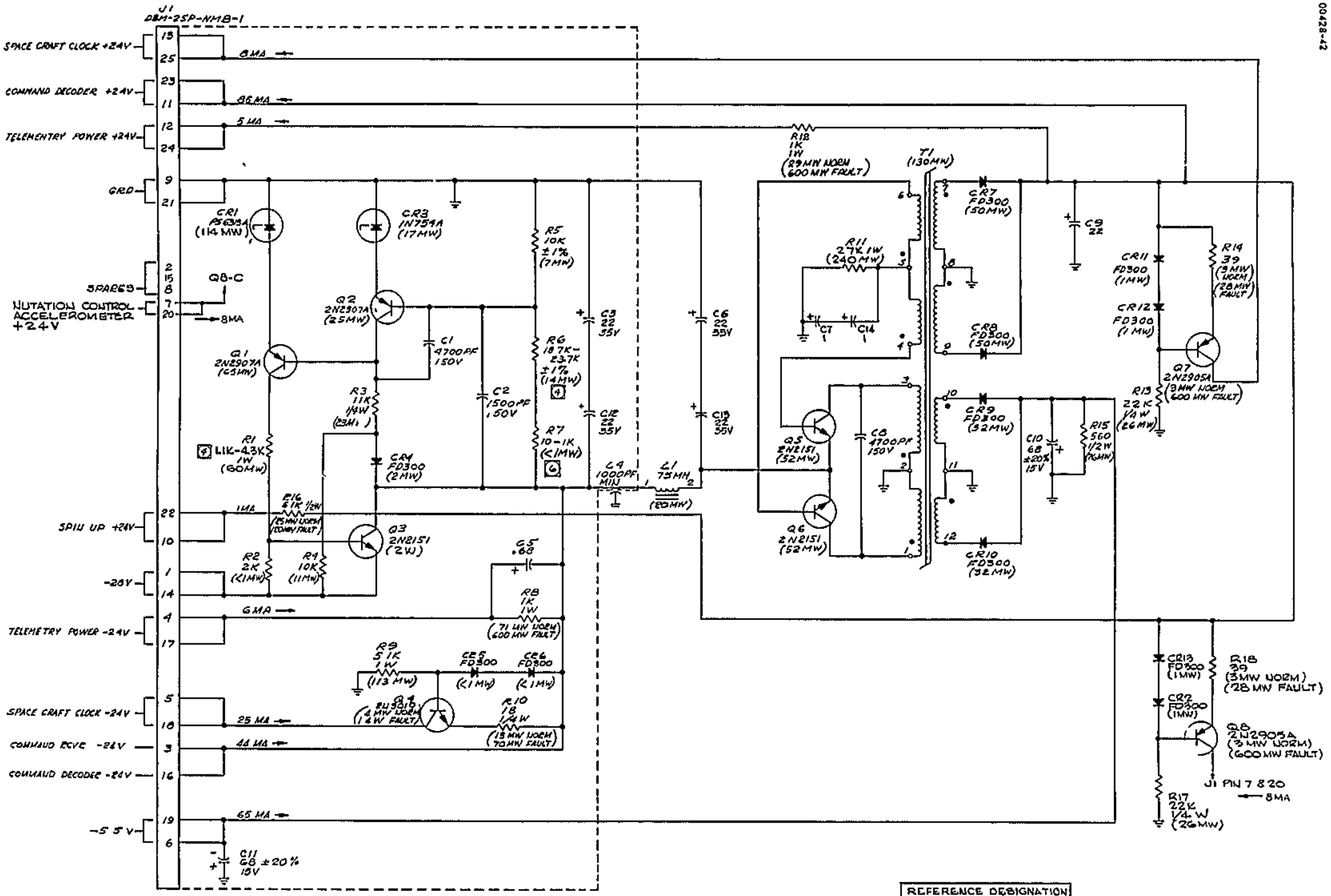
[7] COMPONENT VALUE TO BE SELECTED AT TEST OR MAY BE JUMPER WIRE

[6] COMPONENT VALUE TO BE SELECTED AT TEST

5. DISSIPATIONS SHOWN IN PARENTHESES ARE ACTUAL
[4] COMPONENT VALUE TO BE SELECTED AT TEST OR MAY BE OMITTED
3. RESISTOR VALUES ARE IN OHMS, ±5%, 1/8 WATT
2. CAPACITANCE VALUES ARE IN PICOFARADS ± 10%
1. FOR ASSEMBLY DRAWING AND WIRING DIAGRAM SEE 475156-100

NOTES — UNLESS OTHERWISE SPECIFIED

Figure A-8. TWT Regulator Converter



REFERENCE DESIGNATION	
LAST NO USED	NO NOT USED
CR13	CR14 & UP
C14	C15 & UP
T1	T2 & UP
Q8	Q9 & UP
R18	R19 & UP
L1	L2 & UP

6 COMPONENT VALUE TO BE SELECTED AT TEST OR MAY BE JUMPER WIRE
5 DISSIPATIONS SHOWN IN PARENTHESIS ARE ACTUAL
4 COMPONENT VALUE TO BE SELECTED AT TEST
3 RESISTANCE VALUES ARE IN OHMS, ±5% 1/8W
2 CAPACITANCE VALUES ARE IN MICROFARADS, ±10%, 50V
1 FOR ASSEMBLY DWG & WIRING DIAGRAM SEE 457212-100
NOTES-UNLESS OTHERWISE SPECIFIED

Figure A-9 Command Regulator

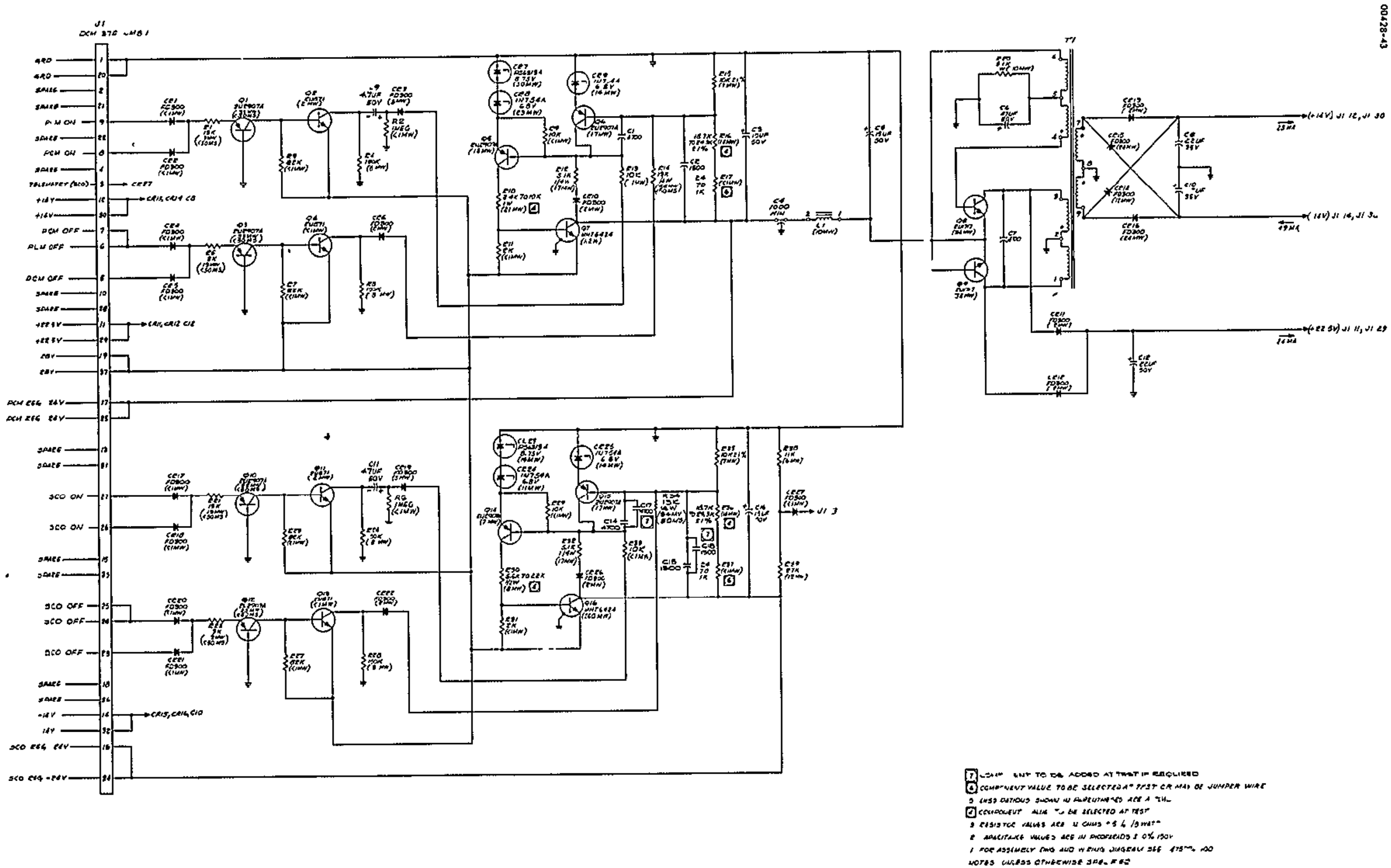


Figure A-10. Telemetry Encoder Regulator

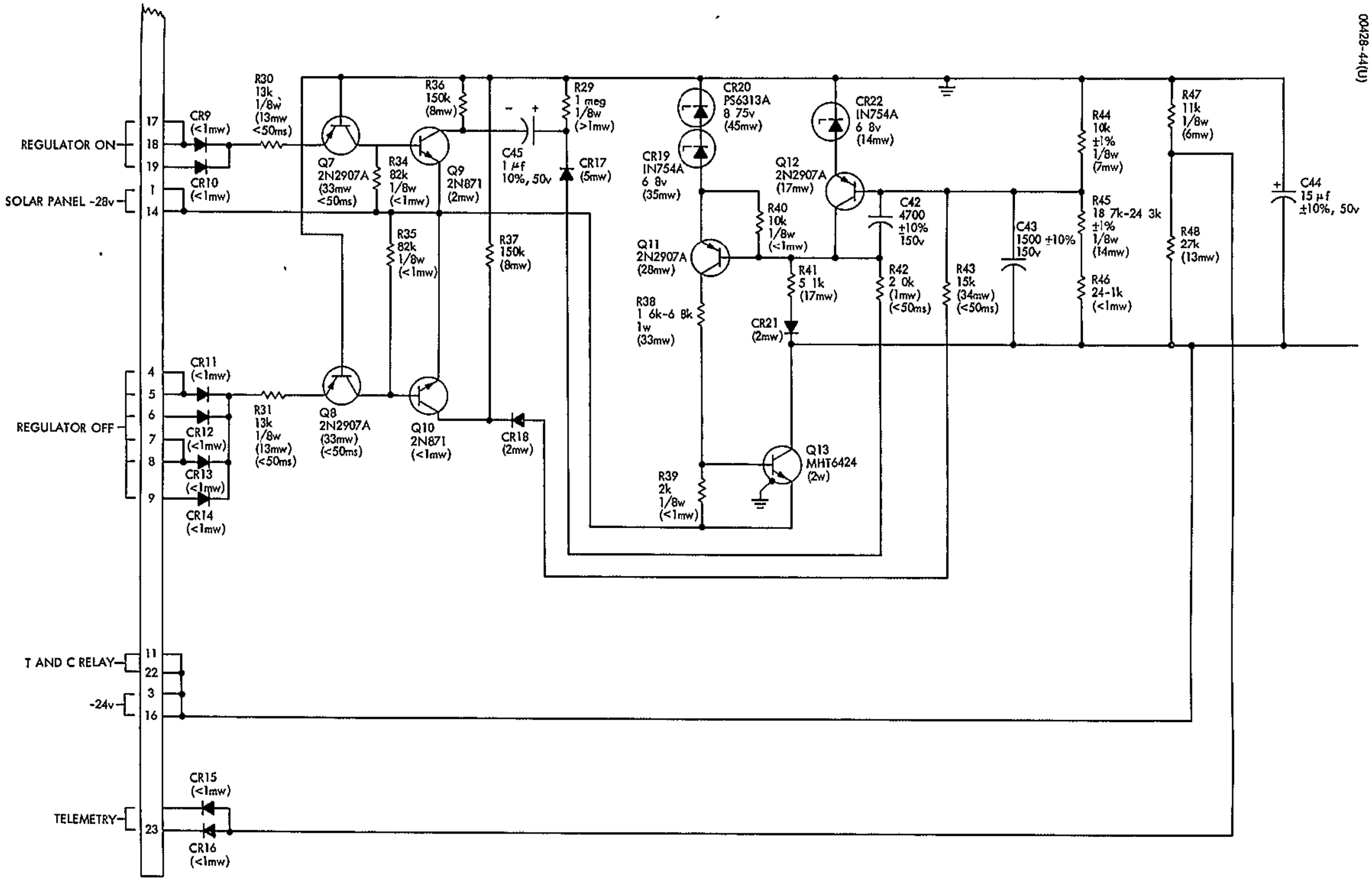


Figure A-11. Telemetry Transmitter Regulator

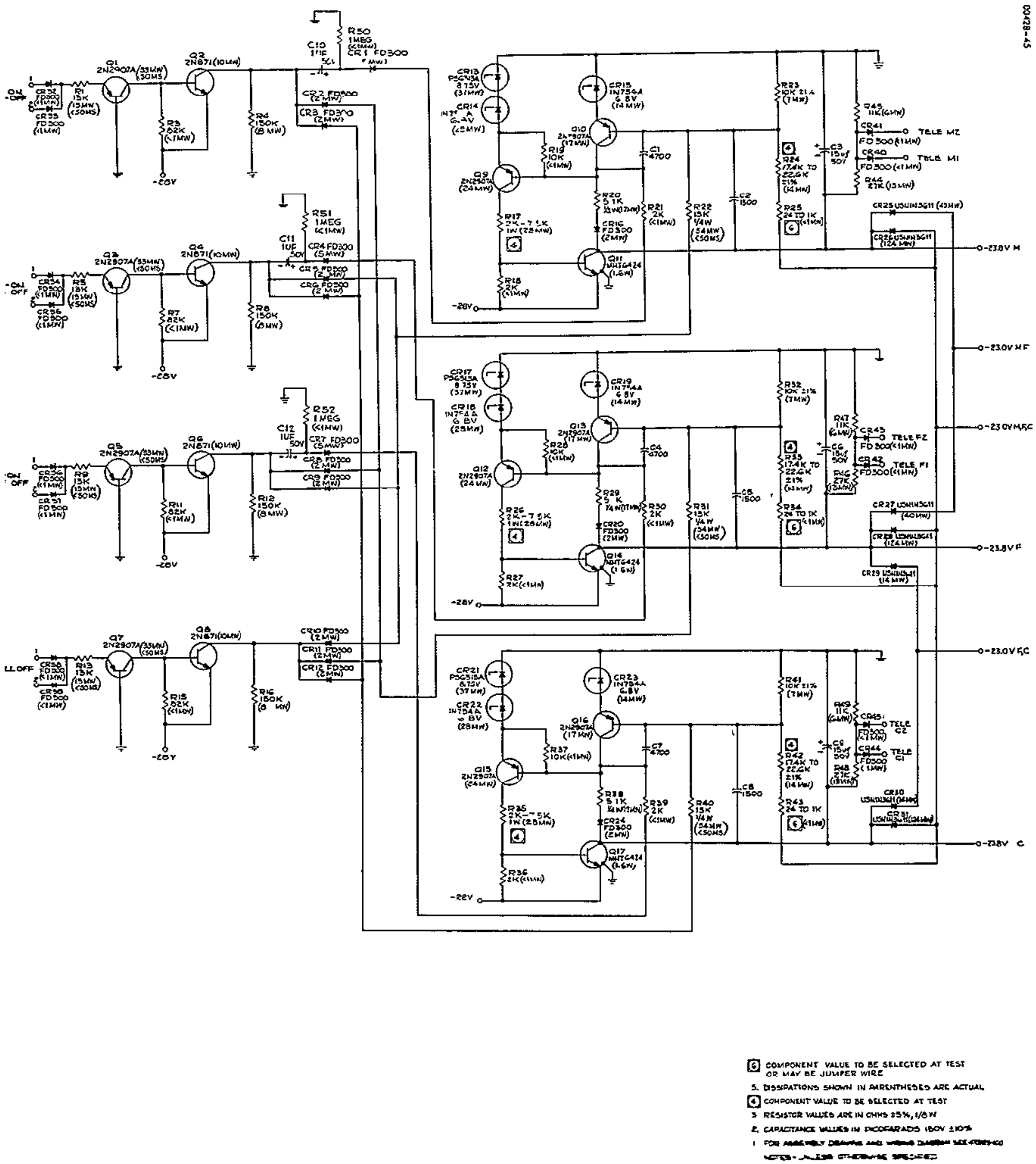


Figure A-12. Transponder Receiver Regulator

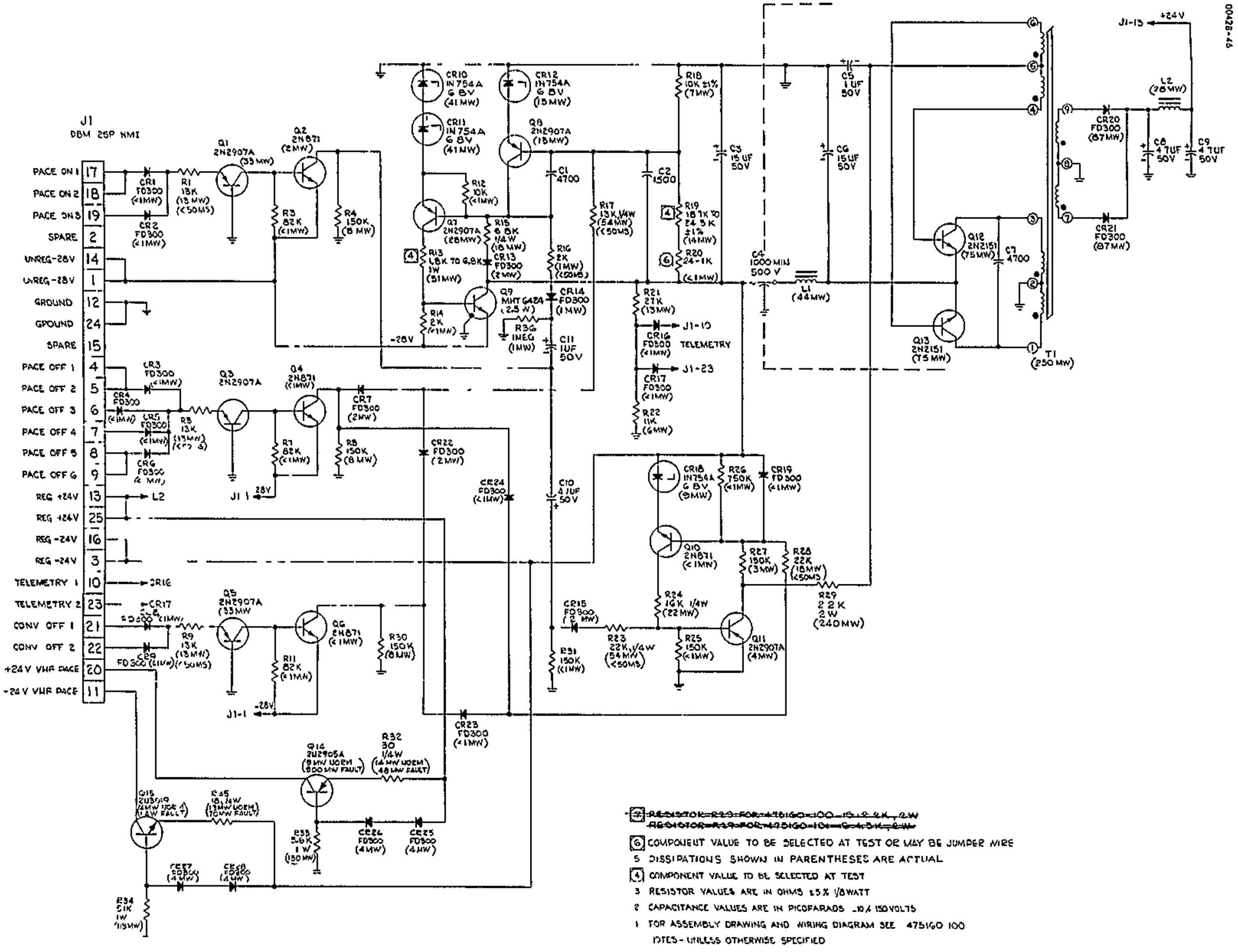
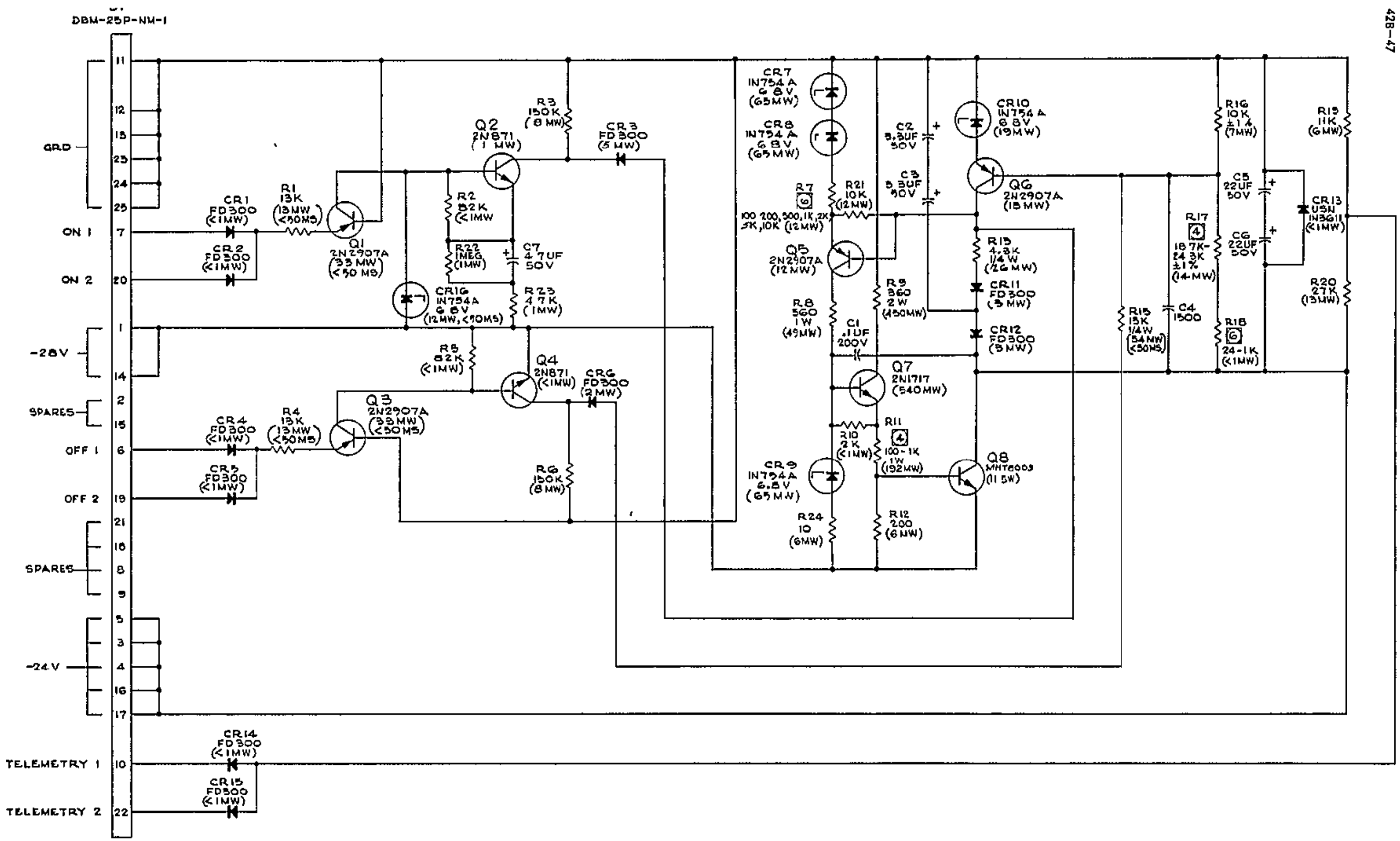


Figure A-13. PACE Regulator



6 COMPONENT VALUE TO BE SELECTED AT TEST OR MAY BE JUMPER WIRE
5 DISSIPATIONS SHOWN IN PARENTHESES ARE ACTUAL
4 COMPONENT VALUE TO BE SELECTED AT TEST
3 RESISTOR VALUES ARE IN OHMS ±5%, 1/8 W
2 CAPACITANCE VALUES ARE IN PICOFARADS, ±10%, 150V
1. FOR ASSY DRAWING AND WIRING DIAGRAM SEE 475308-100 OR 3030341
NOTES — UNLESS OTHERWISE SPECIFIED

Figure A-14. Payload Regulator

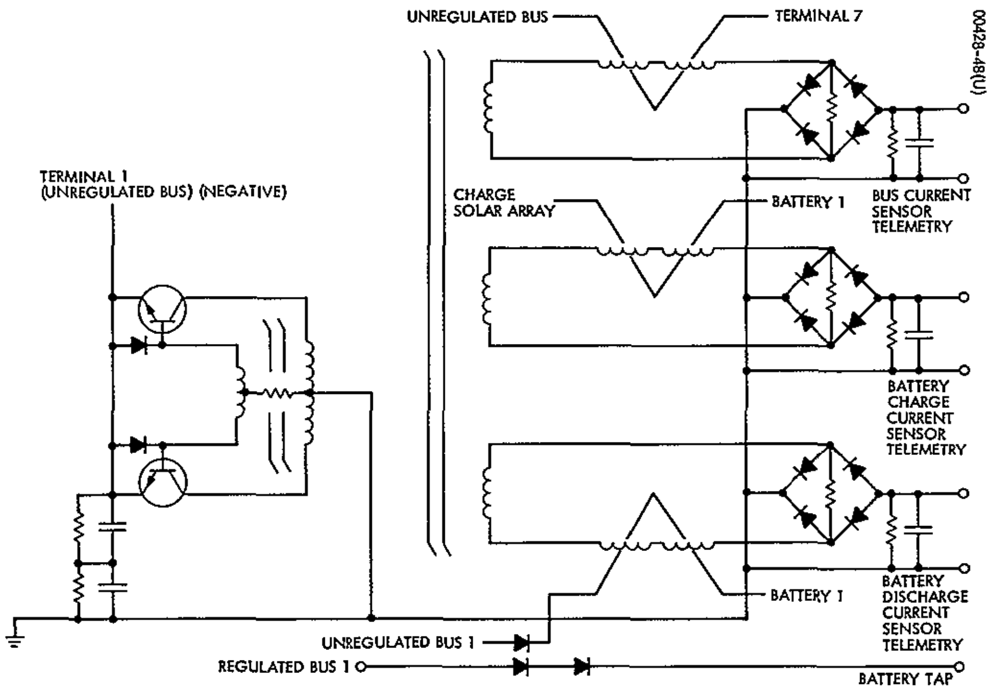


Figure A-15. DC Centralized Regulation System Battery Discharge Unit

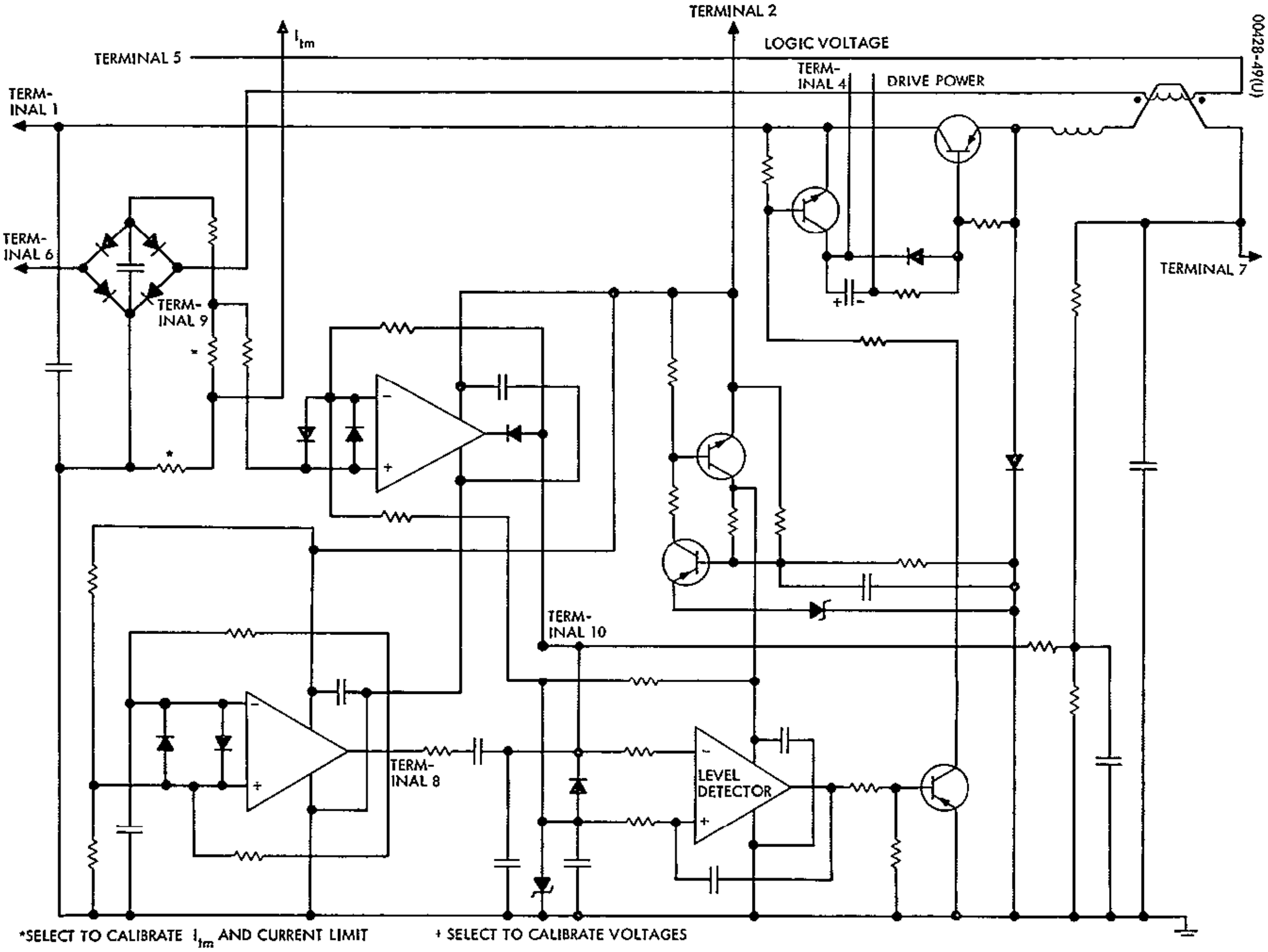


Figure A-16. DC Centralized Regulation System Switching Regulator

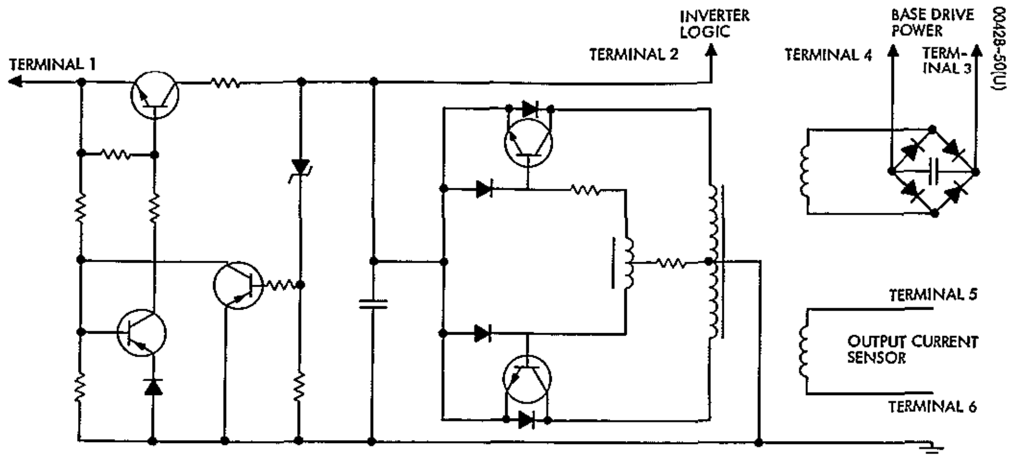


Figure A-17. DC Centralized Regulation System Switching Regulator - Bias Power Supply

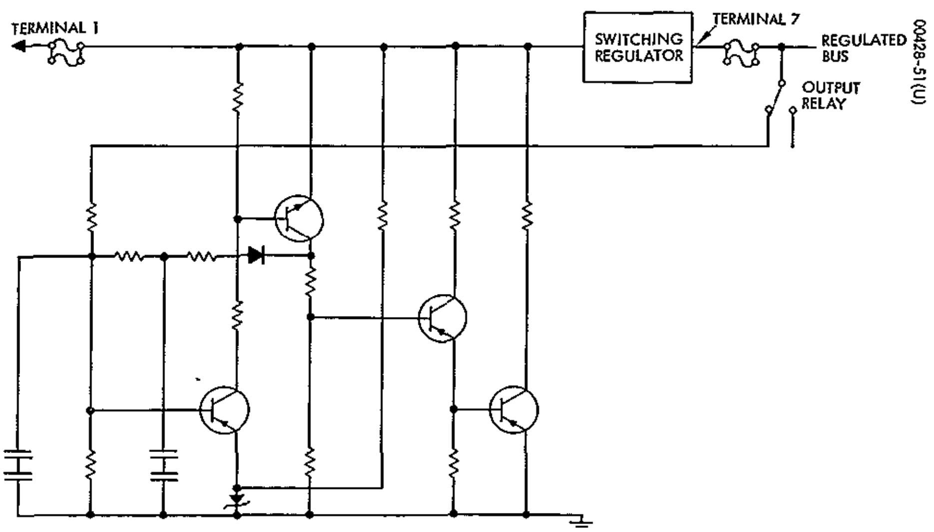


Figure A-18 DC Centralized Regulation System Switching Regulator - Crowbar Circuit

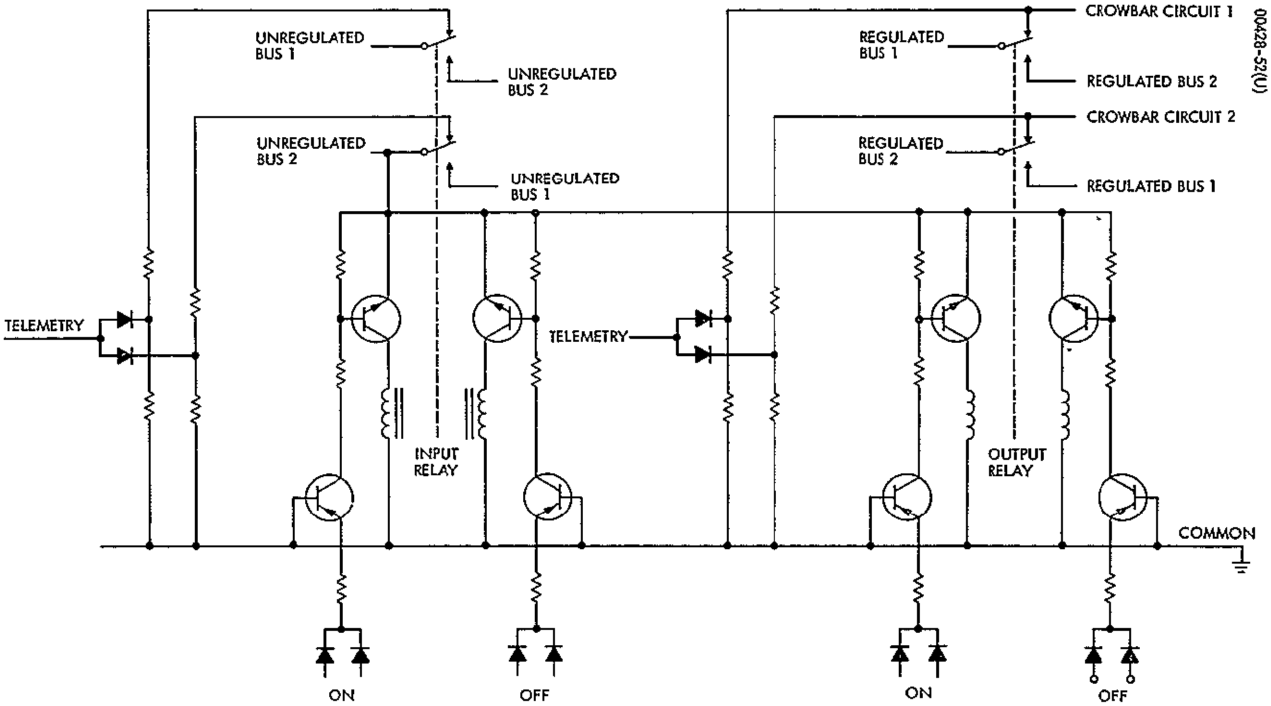


Figure A-19. DC Centralized Regulation System Input/Output Relays

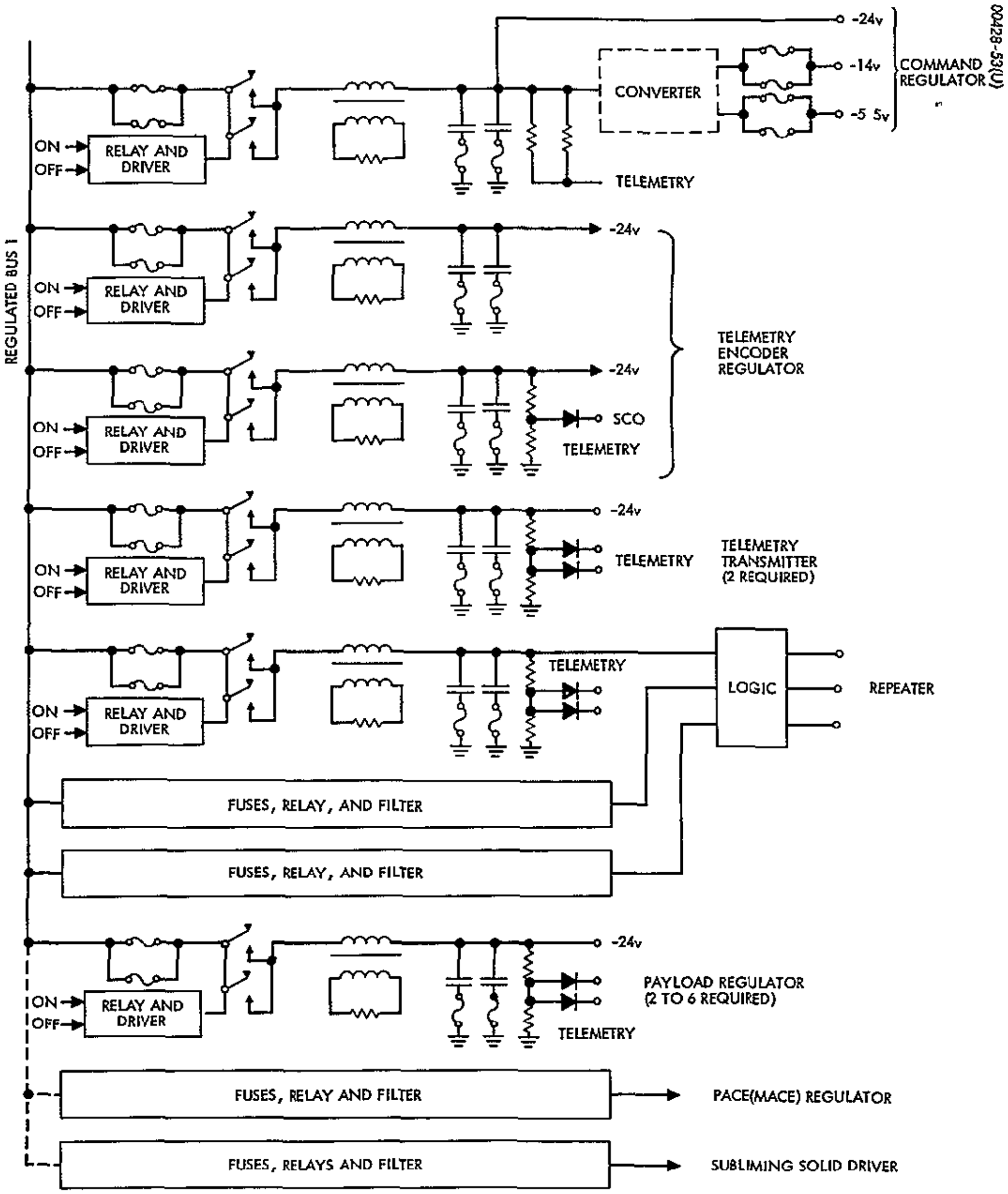


Figure A-20. DC Centralized Regulation System Load Relays and Filters

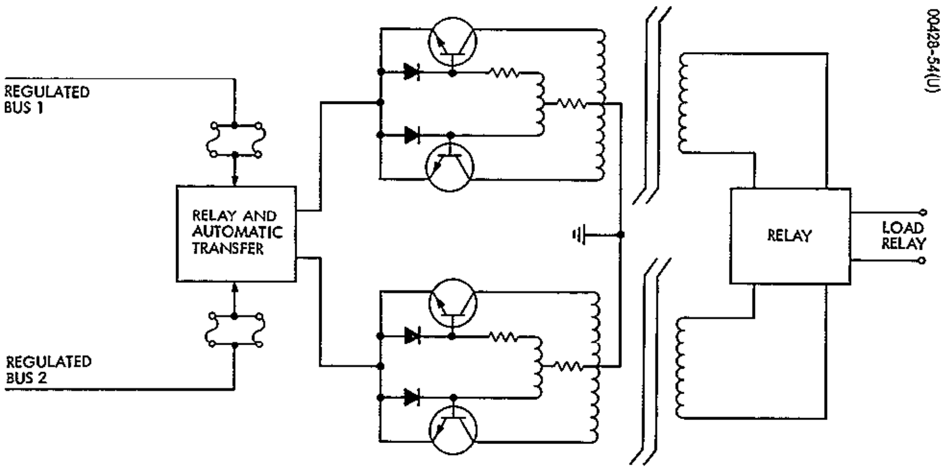


Figure A-21. Redundant Inverters for Centralized Conversion System

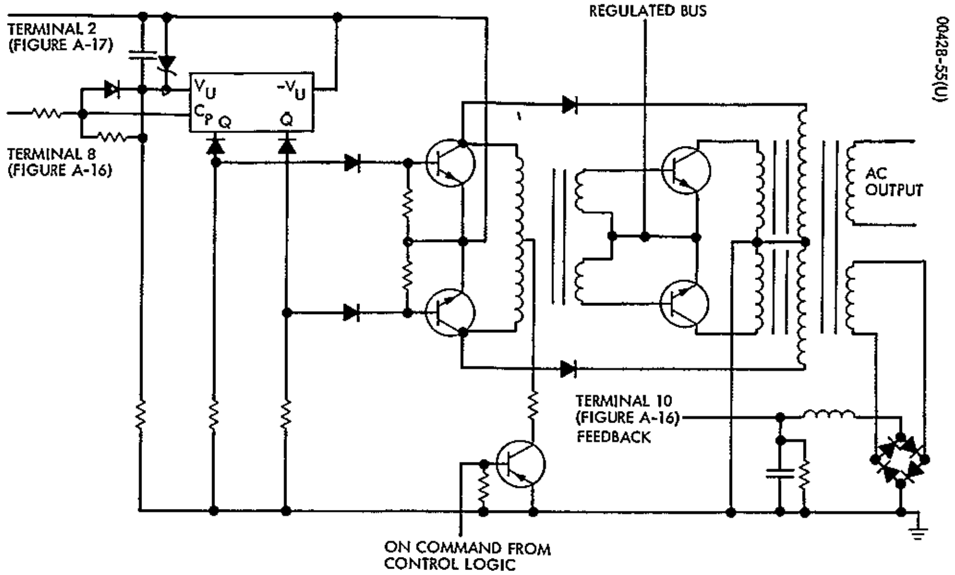


Figure A-22. Centralized AC Distribution Design - Inverter Circuit

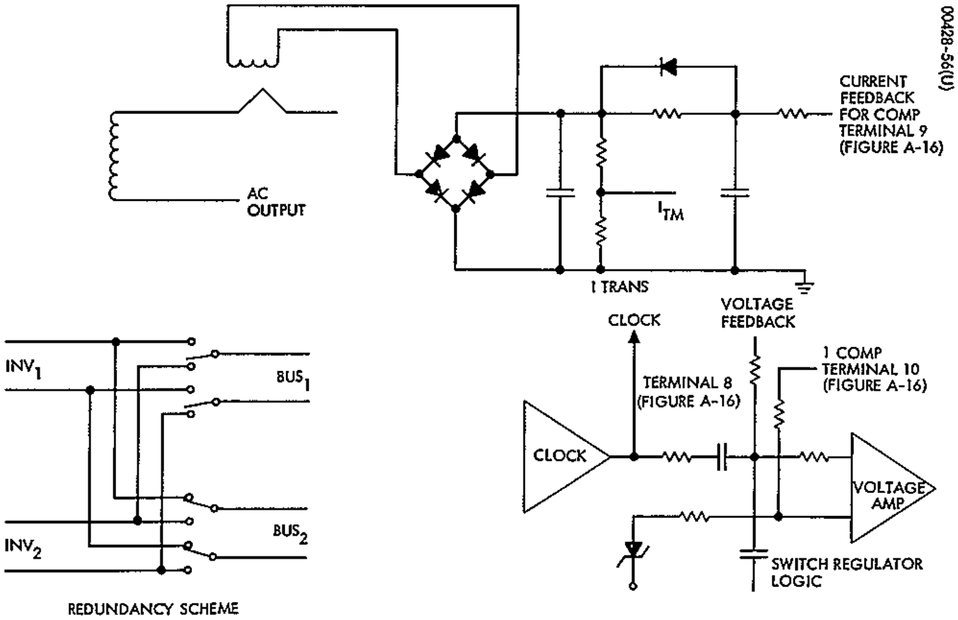


Figure A-23. Centralized AC Distribution Design - Sensing Circuit

APPENDIX B

DESIGN DATA

TABLE B-1. DECENTRALIZED POWER SUBSYSTEM RELIABILITY BLOCK FAILURE RATES

Reliability Block No	Item Name	Item Part No.	Item Failure Rate %/1000 Hours	No. of Items Per Block	Reliability Block Failure Rate %/1000 Hours	Source	3 Years λt	Reliability (t = 3 Years)
1	Bus Voltage Limiter	475270-100	0.0322	4	0.1288	1	0.03385	0.96615
2	Repeater	475129-100	0.0753	1	0.0753	2	0.01979	0.98021
3	Telemetry Transmit Regulator	475220	0.0254	1	0.0254	1 & 2	0.00667	0.99333
4	Telemetry Encoder Regulator (Exclude Converter)	475224-100	0 0359	1	0 0359	1	0 00943	0.99057
5	PACE Regulator (Exclude Converter)	475160-100	0.0402	1	0.0402	2	0.01056	0 98944
6	Command Regulator (Exclude Converter)	475212-100	0.0174	1	0.0174	1	0.00457	0.99543
7	Payload Regulator	475308-100	0.0860	6	0 5160	1	0.06780 (50% Duty Cycle)	0.93220
8	Current Sensor	475271-100	0 0104	4	0.0416	1		
9	Bus Relay	475272-100	0.0405	1	0.0405	1	0.01064	0.98936
10	Discharge Control		0 0862	2	0.1724	1	0.02265	0.97735

SOURCES:

1 - Reference 2 unit failure rates.

2 - Predictions are included in this IDC based upon Reference 2 Part failure rates (see Table B-5)

Block No	Item Name	Item Failure Rate %/1000 Hours	No. of Items Per Block	Reliability Block Failure Rate	3 Year λt	Block Reliability (t = 3 years)
1	Discharge Diodes (Shorted)	0 0045	2	0.0090	0 002365	0.997635
2	4 Diodes (Open)	0 0005	4	0 0020	0.000526	0.999474
3	Input Relay and Driver	0.0146	1	0 0146	0.003837	0 996163
4	Switching Regulator	0.0850	1	0 0850	0.022338	0.977662
5	Crowbar	0.0193	1	0.0193	0 005072	0.994928
6	Repeater Load Relay and Driver (3 Sets) - See Block No 3 for parts	0.0167 (Per Relay and Driver)	3	0 0501	0.01316	0 98684
7	Payload Relay and Driver (6 Sets) - See Block No. 3 for parts	0.0167 (Per Relay and Driver)	6	0.0501 (Duty Cycle = 0 5)	0.01316	0.98684
8	Telemetry Encoder Relay and Driver - See Block No 3 for parts	0.0167 (Per Relay and Driver)	2	0 0334	0.008778	0.991222
9	Telemetry Transmitter Relay and Driver - See Block No 3 for parts	0.0167 (Per Relay and Driver)	1	0.0167	0.00439	0.99561
10	PACE Load Relay and Driver - See Block No 3 for parts	0 0167 (Per Relay and Driver)	1	0.0167	0.00439	0.99561
11	Command Load Relay Driver - See Block No. 3 for parts	0.0167 (Per Relay and Driver)	1	0.0167	0.00439	0.99561
12	Current Sensor	0.0104	6	0.0624		

λ= failure rate

t= operating time

Block No.	Block Name	Part Type	Part Failure Rate, %/1000 Hours	No. of Parts	Total Failure Rate, %/1000 Hours
1	Discharge Diodes (Shorted Mode)	Diode (Shorted)	0.0045	2	0.0090
2	4 Diodes (Open)	Diode (Open)	0 0005	4	0.0020
3	Relay, Driver, and Filter	Fuse	0.02	2 (redundant)	≈0
		Choke	0.002	1	0.002
		Diode, Switch	0.0002	4	0.0008
		Resistor, Carbon	0 0001	9	0.0009
		Transistor, npn, Switch	0 0005	2	0.0010
		Transistor, pnp	0.0020	2	0.0040
		Relay Coil	0.0020	2	0.0040
		Relay			0 0040
		TOTAL	(Relay and Driver) (Relay, Driver and Filter)		0 0146 0.0169
4	Switching Regulator	Diode, Switch	0.0002	8	0.0016
		Resistor, Carbon	0.0001	28	0.0028
		Diode, General Purpose	0.0005	4	0 0020
		Transistor, pnp	0.0020	3	0.0060
		Transistor, npn	0.0005	2	0.0010
		Capacitor, Tantalum	0.0020	7	0.0140
		Capacitor, Ceramic	0 0005	3	0.0015
		Capacitor, Polycarbon	0.0005	1	0.0005
		Diode, Zener	0.0020	2	0.0040
		Integrated Circuit - Digital	0 0020	2	0.0040
		Integrated Circuit - Analog	0.0084	1	0.0084
		SUBTOTAL			0.0458

Table B-5 (continued)

Block No	Block Name	Part Type	Part Failure Rate, %/1000 Hours	No. of Parts	Total Failure Rate, %/1000 Hours
4	Switching Regulator - Power Supply				
		Diode, Power Rectifier	0.0050	4	0.0200
		Diode, General Purpose	0.0005	5	0.0025
		Diode, Zener	0.0020	1	0.0020
		Transistor, npn	0.0005	3	0.0015
		Transistor, pnp	0.0020	2	0.0040
		Resistor, Carbon	0.0001	7	0.0007
		Resistor, Wirewound	0.0005	1	0.0005
		Transformer	0.0020	1	0.0020
		Coil	0.0020	1	0.0020
		Capacitor, Tantalum	0.0020	2	0.0040
		SUBTOTAL			0.0392
		TOTAL REGULATOR			0.0850
5	Crowbar				
		Diode, General Purpose	0.0005	1	0.0005
		Transistor, npn	0.0005	1	0.0005
		Transistor, pnp, Power	0.0020	2	0.0040
		Transistor, pnp	0.0020	1	0.0020
		Diode, Zener	0.0020	1	0.0020
		Capacitor, Tantalum	0.0020	4	0.0080
		Resistor, Carbon	0.0001	7	0.0007
		Resistor, Precision	0.0003	2	0.0006
		Resistor, Power, Wirewound	0.0005	2	0.0010
		TOTAL			0.0193

TABLE B-4. OTHER UNIT RELIABILITY PREDICTIONS

Part	Failure Rate, %/1000 Hours	No. of Parts	Total Failure Rate, %/1000 Hours
475212-100 COMMAND REGULATOR CONVERTER ONLY			
NPN, Switch - Transistor	0.0005	2	0.0010
PNP, All - Transistor	0.002	2	0 0040
Diode, Switch	0.0002	4	0.0008
Diode, General Purpose	0.0005	4	0.0020
Transformer	0.002	1	0.0020
Resistor, Comp	0.0001	7	0.0007
Capacitor, Tantalum	0 0020	4	0.0080
Capacitor, Ceramic	0 0005	1	0.0005
	TOTAL		0.0190
475224-100 PCM ENCODER (TELEMETRY ENCODER) CONVERTER ONLY			
NPN, Switch - Transistor	0.0005	2	0.0010
Diode, Switch - Transistor	0.0002	6	0.0012
Capacitor, Tantalum	0 0020	4	0.0080
Capacitor, Ceramic	0 0005	2	0.0010
Resistor, Carbon	0.0001	1	0 0020
Transformer	0 0020	1	0.0020
	TOTAL		0.0133
475220-201 SERIES REGULATOR ONLY			
NPN, Switch - Transistor	0.0005	3	0 0015
PNP, All - Transistor	0.002	4	0.0080
Diode, Zener	0.002	3	0 0060
Diode, Switch	0.0002	8	0.0016
Diode, General Purpose	0.0005	3	0.0015
Capacitor, Tantalum	0 0020	2	0 0040
Capacitor, Ceramic	0.0005	2	0.0010
Resistor, Comp	0.0001	18	0.0018
	TOTAL		0.0254

Table B-4 (continued)

Part	Failure Rate, %/1000 Hours	No. of Parts	Total Failure Rate, %/1000 Hours
475129-100 REGULATOR, TRANSPONDER RECEIVER			
Transistor, General Purpose, npn	0.0010	7	0.0070
Transistor, pnp	0.0020	10	0.0200
Diode, Switch	0.0002	17	0.0034
Diode, General Purpose	0.0005	15	0.0075
Diode, Zener	0.0020	9	0.0180
Resistor, Carbon	0.0001	44	0.0044
Capacitor, Tantalum	0.0020	6	0.0120
Capacitor, Ceramic	0.0005	6	0.0030
	TOTAL		0.0753
475160-200 REGULATOR, PACE (WITH CONVERTER REMOVED)			
Transistor, General Purpose, npn	0.0010	5	0.0050
Transistor, pnp	0.0020	7	0.0140
Diode, Switch	0.0002	21	0.0042
Diode, Zener	0.0020	4	0.0080
Resistor, Carbon	0.0001	30	0.0030
Capacitor, Ceramic	0.0005	2	0.0010
Capacitor, Tantalum	0.0020	3	0.0060
	TOTAL		0.0402
475212-100 COMMAND REGULATOR EXCLUDING CONVERTER			Data Source
Regulator + Converter	0.0364		
Converter	0.0190		
Regulator (Excluding Converter)	0.0174		
475224-100 PCM ENCODER (TELEMETRY ENCODER) EXCLUDING CONVERTER			
Regulator + Converter	0.0492		Reference 1
PCM Encoder Converter	0.0133		
Regulator (Excluding Converter)	0.0359		

TABLE B-5. SUMMARY OF PART FAILURE RATES

Part Type	Failure Rate, %/1000 hours	Data Source
Bellows - Actuator	0.0082	Test Experience
Sensor	0.0072	Test Experience
Capacitors - Average Satellite	0.0007	R22-100 DC
- Ceramic	0.0005	R22-100 DC
- Glass - Mica	0.0005	R22-100 DC
- Mylar	0.0010	R22-100 DC
- Paper	0.0050	R22-100 DC
- Tantalum	0.0020	R22-100 DC
Connectors - Average Satellite	0.0025	R22-100 DC
Crystals	0.0040	R22-100 DC
Diodes* - Average Satellite	0.0013	R22-100 DC
- Power Rectifier	0.0050	R22-100 DC
- Mixer	0.0200	R22-100 DC
- Switching	0.0002	R22-100 DC
- General Purpose	0.0005	R22-100 DC
- SCR	0.0050	R22-100 DC
- Tunnel	0.0200	R22-100 DC
- Zener	0.0020	R22-100 DC
- Varactor	0.0200	R22-100 DC
Magnetic Components - Average Satellite (Chokes, Coils, Transformers, Inductors)	0.0020	R22-100 DC
Resistors - Average Satellite	0.00013	R22-100 DC
- Carbon Composition	0.0001	R22-100 DC
- Film	0.0002	R22-100 DC
- Power Wirewound	0.0005	R22-100 DC
- Precision Wirewound	0.0003	R22-100 DC
Solar Cells	0.0015	IDC 2861/2207.1/126
Springs	0.0105	AVCO
Transistors - Average Satellite	0.0010	R22-100 DC
- Switching, npn	0.0005	R22-100 DC
- All pnp	0.0020	R22-100 DC
- General Purpose npn	0.0010	R22-100 DC
- Power npn	0.0100	R22-100 DC
- FET	0.0100	R22-100 DC
TWT	0.2000	R22-100 DC
Solder and Weld Connections	0.00002	R22-100 DC
Valve, Control	0.1900	AVCO
Integrated Circuit - Analog	0.0084	SSD P.E. Handbook 06-0203
Digital	0.0020	SSD P.E. Handbook 06-0203
Relay	0.0040	

*The ratio of diode shorts to opens is 9 to 1.

BLE B-6. DECENTRALIZED CONVERTER SUBSYSTEM RELIABILITY PREDICTION

Part Name	Failure Rate, %/1000 Hours	No. of Parts	Total Failure Rate, %/1000 Hours
475160-200 PACE Regulator - dc to dc Converter Only			
Transformer	0 0020	1	0.0020
Diode, Power Rectifier	0 0050	2	0.0100
Capacitor, Tantalum	0.0020	2	0.0040
Choke	0.0020	1	0.0020
Transistor, General Purpose, npn	0.0010	1	0.0010
Transistor, pnp	0.0020	1	0.0020
Capacitor, Tantalum	0.0020	2	0.0040
Choke	0.0020	1	0.0020
Capacitor, Mica	0.0005	1	0 0005
TOTAL			0.0275

TABLE B-7. CENTRALIZED CONVERTER ATS-B SUBSYSTEM RELIABILITY PREDICTI

Part Number	Failure Rate %/1000 Hours	No. of Parts	Total Failure Rate, %/1000 Hours
50 VOLT RMS INVERTER			
Relay	0.0040	1	0.0040
Transistor, npn, General Purpose	0.0010	2	0.0020
Diode, Switching	0 0002	2	0.0004
Transformer	0 0020	1	0 0020
Resistor, Carbon	0.0001	2	0.0002
TOTAL			0.0086
COMMAND STEPDOWN TRANSFORMER AND RECTIFIERS			
Transformer	0.0020	1	0.0020
Diode, Rectifier	0.0050	8	0.0400
Capacitor, Tantalum	0.0020	2	0.0040
Transistor, pnp	0.0020	2	0.0040
Resistor, Carbon	0.0001	5	0.0005
TOTAL			0.0505
TELEMETRY STEPDOWN TRANSFORMER AND RECTIFIERS			
Transformer	0.0020	1	0.0020
Diode, Rectifier	0.0050	6	0.0300
Capacitor, Tantalum	0.0020	3	0.0060
TOTAL			0.0380
PACE STEPDOWN TRANSFORMER AND RECTIFIERS			
Transformer	0.0020	1	0.0020
Diode, Rectifier	0 0050	2	0.0100
Choke	0 0020	1	0 0020
Capacitor, Tantalum	0.0020	2	0 0040
TOTAL			0.0180
TRANSFER RELAY			
Transistor, npn, switching	0.0005	5	0.0025
Diode, Switching	0.0002	8	0.0016
Resistor, Carbon	0.0001	15	0.0015
Capacitor, Tantalum	0.0020	2	0.0040
Capacitor, Ceramic	0.0005	2	0.0010
TOTAL			0.0106